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MISSION / SYSTEM REQUIREMENTS AND ANALYSES

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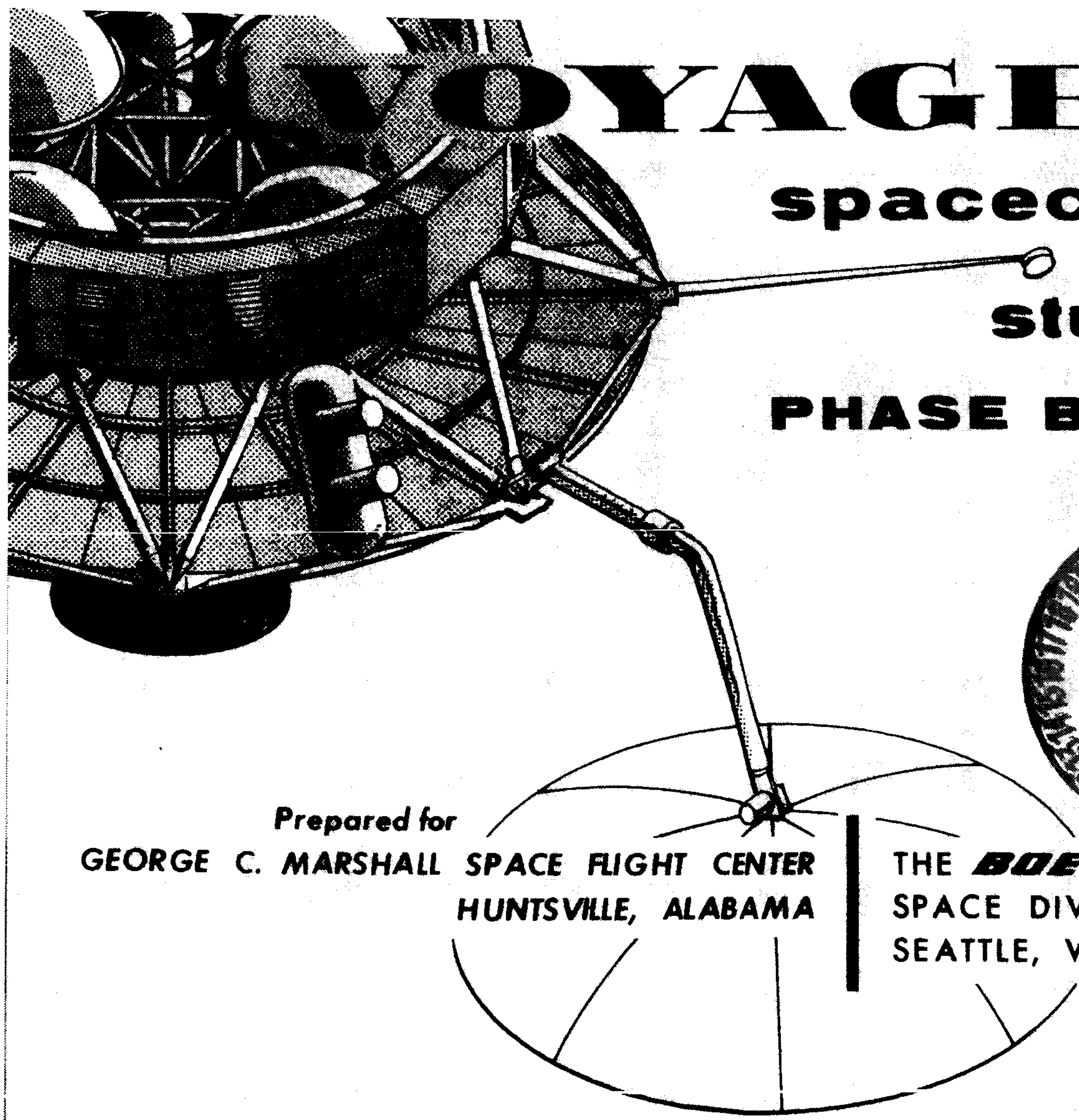

VOYAGER

spacecraft system studies

PHASE B, TASK D

Prepared for
GEORGE C. MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA

THE BOEING COMPANY
SPACE DIVISION
SEATTLE, WASHINGTON

VOYAGER

SPACECRAFT SYSTEM STUDY

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FINAL TECHNICAL REPORT

PHASE B, TASK D

VOLUME II
MISSION/SYSTEM REQUIREMENTS AND ANALYSES

D2-115002-2

OCTOBER 1967

Prepared For:

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

GEORGE C. MARSHALL SPACE FLIGHT CENTER

HUNTSVILLE, ALABAMA

UNDER

CONTRACT NO. NAS8-22602

THE BOEING COMPANY • SPACE DIVISION • SEATTLE, WASHINGTON

FOREWORD

This series of documents summarizes the work performed under the George C. Marshall Space Flight Center contract, NAS 8-22602 entitled, "Voyager Spacecraft System, Phase B, Task D." The work was performed over the period June 16 through October 16, 1967.

The contracted work consisted of engineering studies leading to a definition of a Voyager Mars/Spacecraft system capable of performing the 1973 mission. To ensure flexibility of design, additional analyses were conducted to determine the adaptability of the 1973 spacecraft to perform the 1975-1977-1979 Mars missions. The 1973 flight spacecraft definition was used to identify the operational support equipment including mission-dependent equipment requirements and the software necessary to satisfactorily conduct the 1973 mission operations. Logistics considerations were identified for the 1973 system from point of manufacture through launch operations.

The contract also required the completion of five selected engineering tasks that were designed to highlight key areas and lead to specific conclusions and recommendations.

The detailed results of the contracted work is contained in the following reports:

• Summary Report	Volume I D2-115002-1
• Mission/System Requirements and Analyses	Volume II D2-115002-2
• Spacecraft Functional Description	Volume III D2-115002-3
• Selected Engineering Tasks	Volume IV D2-115002-4

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CONTENTS

1.0 INTRODUCTION AND SUMMARY

1.0 INTRODUCTION AND SUMMARY

This document is the second volume of the Phase B, Task D Final Technical Report on Voyager spacecraft system studies performed by The Boeing Company. These studies were conducted for the George C. Marshall Space Flight Center under NASA Contract No. NAS8-22602. The data contained in this document include mission and system requirements and analyses for the Mars Voyager missions.

1.1 BACKGROUND AND OBJECTIVES

Previous Voyager spacecraft system studies resulted in the analysis, definition, and functional description of a Voyager flight spacecraft capable of performing a 1971 Mars mission. The spacecraft definition had to be revised to reflect (1) a change in the first launch opportunity from 1971 to 1973, and (2) a need to develop a spacecraft system that can be adapted to the 1975, 1977, and 1979 missions with minimum modifications.

To define the flight spacecraft for the 1973 mission, it was necessary to establish applicable performance and design requirements that also will allow maximum use of existing 1971 designs in lieu of new designs. Consequently, the first objective of the studies reported in this volume was to compare and identify common 1971 and 1973 mission and system requirements. The degree of commonality of these requirements will indicate the degree to which existing 1971 spacecraft concepts and designs could satisfy the 1973 mission.

The second objective of the studies was to develop a set of performance and design requirements for the 1973 spacecraft mission. This set consists of common 1971-1973 requirements as well as requirements unique to the 1973 mission. The set of 1973 mission requirements is necessary to the revised spacecraft definition effort.

The third objective of the studies was to develop preliminary spacecraft performance and design requirements for the subsequent Mars Voyager missions in 1975, 1977, and 1979. These requirements, when reflected on the 1973 spacecraft design, would ensure that the resulting design could be adapted to these missions with a minimum of modifications.

Preliminary mission and spacecraft performance and design requirements are provided in project source documentation. These preliminary requirements must be assessed. Such assessments may require systems analyses and tradeoffs. Therefore, the fourth objective of the studies was to perform spacecraft system analyses and trade studies.

After the establishment of spacecraft performance and design requirements, definition and description of the spacecraft can be accomplished. With the spacecraft thus defined, the requirements associated with other spacecraft system elements can then be established. Such spacecraft system elements include OSE and MDE. These elements are important because (1) their cost is a considerable fraction of total program cost, and (2) they contribute to the probability of mission success. Therefore, the development of preliminary OSE and MDE requirements and concepts is the fifth objective of the studies documented in this volume.

The sixth and final objective of this portion of the contract was to identify potential Voyager spacecraft logistics problems. This logistics study was motivated by the large size of the Voyager spacecraft. It is anticipated that, because of its size, Voyager could present unusual problems in transportation and handling from point of manufacture to point of launch. These problems, in turn, could lead to unusual OSE requirements, particularly in handling and shipping equipment.

1.2 SUMMARY

1.2.1 Mission and System Requirements

Mission and system performance and design requirements were organized into 15 categories pertaining to the (1) mission, (2) spacecraft, (3) spacecraft subsystems, and (4) interfacing Voyager project elements, e.g., the flight capsule. The comparison between the 1971 and 1973 requirements, as contained in project source documentation, revealed that (1) many of the requirements are similar, (2) few of the requirements are common, and (3) significant differences exist in the areas of orbit lifetime, flight capsule weight, orbit insertion velocity capability, and celestial reference occultation durations. Even so, it was concluded that the 1971 spacecraft concepts satisfy the 1973 spacecraft performance and design requirements.

Subsequent to the comparison of 1971-1973 requirements, the following were developed: (1) a set of requirements for the 1973 mission, and (2) preliminary requirements for the 1975-1977-1979 missions. The 1973 requirements set was sufficiently complete to allow the definition of the 1973 flight spacecraft. Key new features of the 1973 requirements included the following: (1) deletion of the no-Canopus occultation constraint, (2) increase of the allowable duration of solar occultation after the first 30 days in orbit, and (3) target weight and reliability allocations for the flight spacecraft elements. The preliminary 1975-1977-1979 requirements were developed in sufficient depth to allow identification of changes in spacecraft design concepts. Key new features of the 1975-1977-1979 requirements included (1) Type II Mars transfer trajectories in 1975 and 1977, and (2) increased weight and power allocations reflecting primarily a heavier flight capsule and improved spacecraft photoimaging equipment. The impact of the 1975-1977-1979 requirements was evaluated, as these requirements may require spacecraft design concept changes in the telecommunication and attitude control subsystems.

1.2.2 System Analyses and Trade Studies

The 1973 mission profile was examined. Key events were identified, which led to system level analyses and trade studies including the following nine studies.

The objectives of the first study, trajectory selection analysis, were: (1) to establish that the specified baseline orbit satisfies the orbit lifetime requirements and Sun occultation constraint; (2) to assess the capability of the baseline orbit to satisfy the objectives of the photoimaging experiment; and (3) to define the interplanetary transfer trajectories, aiming points, and midcourse correction and orbit trim maneuver requirements that will accommodate the baseline orbit in the 1973-1975-1977-1979 launch opportunities.

The analysis resulted in the following key conclusions:

1) The baseline orbit (500-km periapsis altitude by 18,600-km apoapsis altitude) satisfies the planetary quarantine constraint.

2) The baseline orbit satisfies the Sun occultation constraint.

3) The baseline orbit provides adequate illumination angle at or near periapsis, latitude coverage, and wave-of-darkening coverage for the photoimaging experiment.

4) An orbit insertion ΔV allocation of 1.59 km/sec satisfies the baseline orbit requirement.

5) Orbit trim ΔV allocations for the baseline orbit must be increased from 150 to 210 m/sec to accommodate early arrival dates for the 1973 mission.

6) Three midcourse trajectory corrections will satisfy planetary quarantine and baseline orbit requirements.

7) By allocating the required 8-day arrival time bias maneuver to the two planetary vehicles, i.e., + 4 days for the first, and -4 days for the second, the combined midcourse corrections and biasing maneuver ΔV requirements can be achieved with the 210 m/sec allocated to each planetary vehicle.

The objective of the second study, launch delays, was to define alternate modes for accomplishing a 1973 mission in the event of launch slides beyond the full-capability launch window. The results of this study indicate that, at best, launch period increases in excess of 2 months are feasible, provided only a single flight spacecraft is launched by a Saturn V and a marginal 60-hour Mars orbit is attained.

The objective of the third trade study, mission energy balance, was to determine the maximum attainable weight in Mars orbit. The current Saturn V payload capability was assumed. This study led to the following key conclusions: (1) the maximum payload weight in Mars orbit is sensitive to arrival date and launch period (1973 and 1979 in particular), and (2) project payload contingency and minimum reasonable launch periods rule out Type I trajectories in 1975 and 1977.

The objectives of the fourth trade study, encounter communication distance, was to identify mission modes that minimize Mars-to-Earth communication distance at encounter. Study results indicate that:

1) The greatest communication distance reduction in 1973 (35%) derives from a high-arrival-energy, early-arrival-date mission.

2) A two-step orbit insertion technique allows an additional 6% communication distance reduction. (This is achieved by first deorbiting the capsule from a highly eccentric orbit and then adjusting the orbit).

3) Loading the propellant tanks to their full (1979) capacity results in a further 1973 communication distance reduction of 15%.

The objective of the fifth trade study, capsule canister separation, was to determine the time to separate the forward capsule-sterilization canister during the mission sequence. This study led to the conclusion that the canister could be separated prior to orbit insertion. However, to increase confidence in the maintenance of planetary quarantine, and to simplify flight capsule thermal control, it is recommended that the canister be separated subsequent to orbit insertion.

The objective of the sixth study, propulsion engine selection, was to select a preferred engine concept for Voyager from the following four engine concepts: (1) lunar module descent engine (LMDE), (2) LMDE with a cluster of four C-1 engines, (3) Agena Model 8517, and (4) the Transtage engine. As a result of this study, the LMDE engine was selected as the preferred Voyager engine for all spacecraft propulsion maneuvers.

The objective of the seventh trade study, orbital data rate, was to establish the most reasonable amount of data to be transmitted to Earth during the 1973 Mars orbital mission. The study results indicated that a total data return of 1.57×10^{11} bits during the 180-day orbital mission will enable high resolution (10 meters average) coverage of 0.1% of the planet's surface, and medium resolution (300 meters average) coverage of 75% of the planet's surface. The attendant average data rate is 12,500 bps.

The objective of the eighth trade study, laser telecommunications feasibility, was to evaluate the feasibility of laser telecommunications for Voyager. The results of this study indicated that a laser telecommunication experiment is feasible and, therefore, recommended for the 1975 mission. If successful development is shown, then a change to laser telecommunication in 1977 and 1979 can be implemented.

The objective of the ninth and last trade study reported in this volume, RF high data rate, was to explore the possibility of increasing the information bandwidth of an RF spacecraft-to-Earth data link. This study resulted in the following key conclusions: (1) spacecraft effective radiated power (ERP) can be increased by increasing antenna diameter without tightening spacecraft pointing requirements, and (2) use of an analog VSB-AM/PM carrier modulation technique in the 1977-1979 Voyager mission will result in a 4.5-db (i.e., a factor of 3) improvement over the biorthogonally coded digital phase shift keying/phase modulation (PSK/PM) technique that was selected for the baseline 1973 spacecraft.

1.2.3 Preliminary OSE and MDE Requirements and Concepts

An OSE concept compatible with the 1973 flight spacecraft configuration (described in Volume III of this final technical report) was established. The OSE included launch control equipment (LCE), system test complex (STC), assembly, handling, and shipping equipment (AHSE), and fueling and servicing equipment (FSE). For the established OSE concept, test and operation flows were developed. Preliminary OSE requirements and related test locations were then identified. Interfaces between the Voyager project systems were also identified to ensure OSE compatibility. Next the commonality between the test system OSE and the MDE was established. This avoids equipment duplication.

After the establishment of the above requirements and interfaces, existing Apollo ACE and Saturn ESE concepts were evaluated for applicability to Voyager. It was concluded that Voyager flight spacecraft OSE should use the same basic concept as the Apollo ACE and Saturn ESE, i.e., automated testing involving operator consoles, computers, and special purpose interfacing equipment. Such a system would be mobile, thereby reducing system test equipment impact on schedule, location, and facilities.

1.2.4 Spacecraft Logistics Considerations

Spacecraft level test and operational flows were reviewed. Particular attention was given to assembly and test, site support, and transportation. Problems that could not be solved easily by normal design practices and normal logistics practices were then identified. These problems were segregated into the following three categories: (1) problems involving interfaces between other contractors or Government agencies, (2) equipment availability problems resulting from schedule conflicts with other programs, and (3) common Voyager project-element problems that should have a common solution.

The following eight specific logistics problem areas were identified: (1) spacecraft fueling procedure, (2) payload module integration, (3) payload stack test, (4) cleanliness and contamination, (5) training, (6) payload module malfunction, (7) payload launch operations system interface, and (8) spares considerations. Recommended solutions to all eight problems then were developed.

Detailed discussions of the objective, approach, results, and conclusions for the mission/system requirements and analyses studies are given in the following sections.

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2.0 MISSION AND SYSTEM REQUIREMENTS

2.0 MISSION AND SYSTEM REQUIREMENTS

This section contains the mission and system requirements for the Voyager Mars flight spacecraft design. The requirements were developed primarily for the 1973 mission. Preliminary requirements for the 1975, 1977, and 1979 missions were developed primarily for assessing mission versatility of the 1973 spacecraft design. The steps involved in developing the above requirements are shown in Figure 2-1 and were as follows:

1) A comparison was made of the previously developed 1971 mission requirements and the initial 1973 mission specification to determine the degree of commonality.

2) A compilation of 1973 requirements was made by using (a) data developed in the first step, (b) definition of the natural environment contained in NASA TMX-53616, (c) results of key mission/system analyses and trade studies, and (d) data obtained through coordination with MSFC.

3) Preliminary requirements for the 1975, 1977, and 1979 mission were generated.

4) The gross effect that the 1975, 1977, and 1979 preliminary requirements have on a 1973 spacecraft configuration was determined.

The requirements resulting from the above four steps control the flight spacecraft configuration. Preliminary requirements for operational support equipment (OSE) and mission dependent equipment (MDE) are given in a subsequent section of this volume.

2.1 1971/1973 REQUIREMENT COMPARISON

2.1.1 Purpose

The purpose of this study was to assess the applicability of the 1971 spacecraft concepts to perform the 1973 mission.

2.1.2 Scope

The comparison between missions was established by evaluating the 1971 and 1973 mission requirements. The Phase 1A, Task B Final Report (Boeing Document D2-82709-6) was the data source for the 1971 mission requirements. The 1973 Mission Specification was used as a preliminary source of 1973 requirements. The 1973 mission requirements were subsequently augmented and modified by MSFC guidelines as indicated in Section 2.2.

The requirements for the 1971/1973 launch opportunities were presented and compared. The requirements were organized into 15 major headings. These headings are used as subject matter groupings of applicable parameters. The same organization of data is used in Sections 2.2, 2.3, and 2.4, so that continuity is maintained in all the requirement presentations.

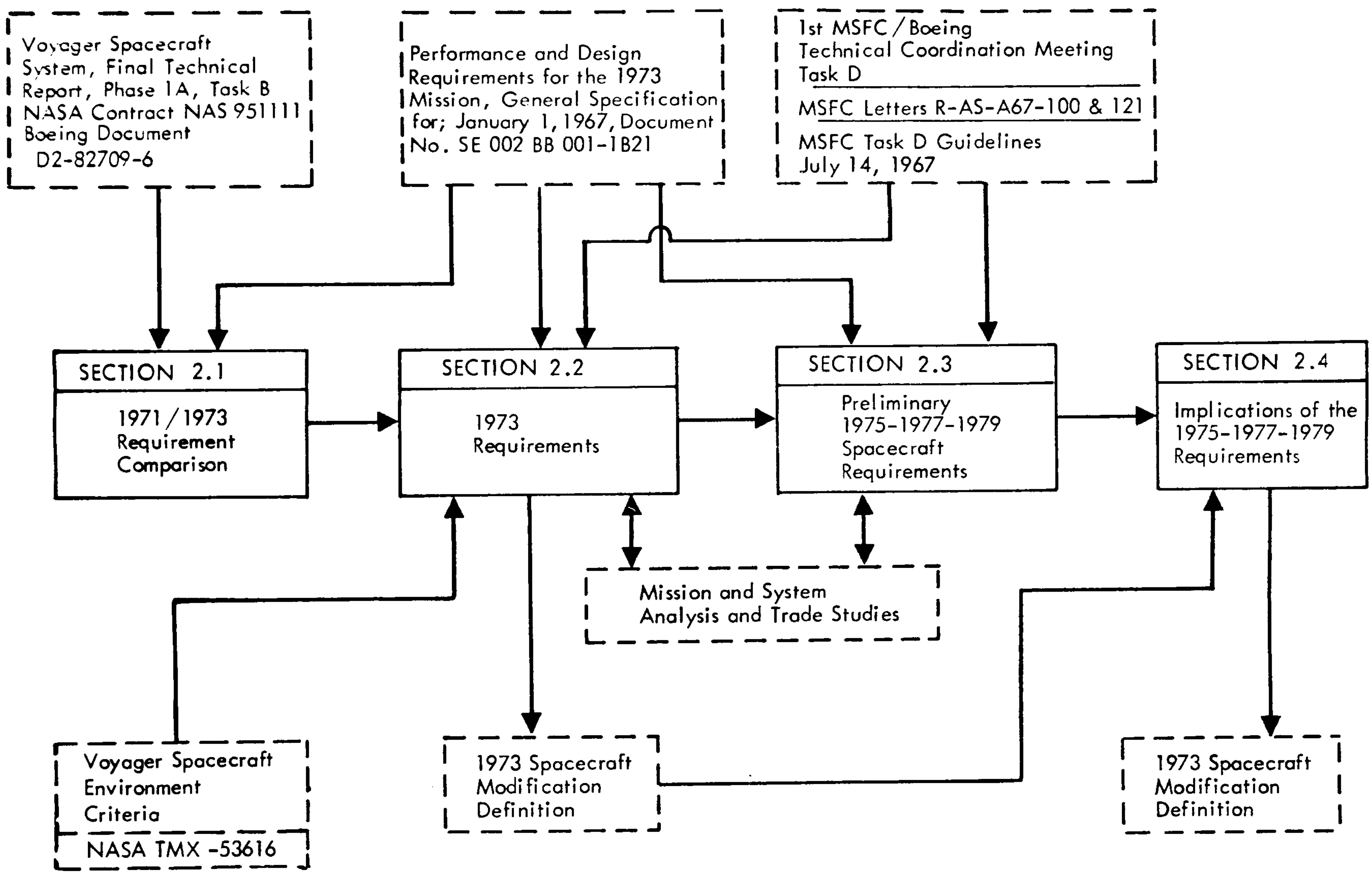


Figure 2-1: APPROACH TO THE DEVELOPMENT OF PERFORMANCE AND DESIGN REQUIREMENTS

The 15 subject-matter headings of the 1971/1973 requirements comparison (see Table 2-1)

- Mission Definition and Profile
- Mission Requirements--Operational Life
- Trajectory Parameters--Interplanetary Transfer
- Trajectory Parameters--Operations Planning
- Trajectory Parameters--Mars Orbit
- Target Weight Allocations
- General Design Requirements
- Propulsion Requirements
- Capsule Support Requirements
- Data Return Requirements
- General Capsule Requirements
- Electrical Power Requirements
- Reliability Allocations
- Spacecraft Stabilization Requirements
- Science Payload Requirements

2.1.3 Conclusions

The comparison was conducted at the mission and system level. The comparison resulted in identifying few common, but many similar requirements. Significant differences were found in the following parameters:

- Orbit lifetime (mission requirements - operational lifetime)
- Flight capsule weight (target weight allocations)
- Orbit insertion velocity (propulsion requirements)
- Celestial occultation times (trajectory parameters - Mars orbit)

Even so, it was concluded that the 1971 spacecraft concepts satisfy the 1973 spacecraft performance and design requirements.

2.2 1973 REQUIREMENTS

2.2.1 Purpose

The aim in this study was to establish a definitive set of requirements for the 1973 Mars Voyager mission.

Table 2-1: COMPARISON OF 1971 AND 1973 INITIAL REQUIREMENTS (Sheet 1 of 4)

REQUIREMENTS	VOYAGER PHASE 1A, TASK B, FINAL REPORT BOEING DOCUMENT D2-82709-6 (1971 Mission)	PERFORMANCE AND DESIGN REQUIREMENTS FOR 1973 VOYAGER MISSION, GENERAL SPECIFICATIONS FOR, JAN 1, 1967
MISSION DEFINITION AND PROFILE		
Flt capsule operational mode	Descent and entry	Descent, entry, and surface
Boost shroud separation technique	Clam-shell design	Over-the-nose design
Initial launch year	1971	1973
Flt S/C operational mode	Orbit	Orbit
MISSION REQUIREMENTS - OPERATIONAL LIFE		
S/C orbit life	1 month minimum 6 months goal	2 months minimum 6 months goal
Life on Mars surface	Nonsurvival	1 day plus time to transmit data
TRAJECTORY PARAMETERS - INTERPLANETARY TRANSFER		
Boost performance guarantee	63,000 lb @ C_3 of 25 km^2/sec^2: 115 degrees azimuth	55,300 lb @ C_3 of 32.5 km^2/sec^2: 115 degrees azimuth
Mars arrival time period	Not specified	2/3/74 - 3/21/74
Earth parking orbit holding time	2 - 90 minutes	10 - 90 minutes
Time following trans-Mars injection for initial trajectory correction and biasing	4 days	2 - 20 days
TRAJECTORY PARAMETER - OPERATIONS PLANNING		
Launch azimuth boundaries	60 - 115 degrees	90 - 115 degrees
Minimum launch period	45 days	30 days
Minimum daily launch window	2 hours	1 hour

Table 2-1: COMPARISON OF 1971 AND 1973 INITIAL REQUIREMENTS (Sheet 2 of 4)

REQUIREMENTS	VOYAGER PHASE 1A, TASK B, FINAL REPORT BOEING DOCUMENT D2-82709-6 (1971 Mission)	PERFORMANCE AND DESIGN REQUIREMENTS FOR THE 1973 VOYAGER MISSION, GENERAL SPECIFICATIONS FOR, JAN 1, 1967
TRAJECTORY PARAMETERS - OPERATIONS PLANNING (continued)		
Absolute DLA limits	-5 deg $\leq$DLA $\leq$-39.5 deg: inclination to ecliptic >0.1 deg	36 deg $\geq$DLA $\geq$5 deg: inclination to ecliptic >0.1 deg
Maximum VHP (Mars encounter)	4.5 km/sec	3.25 km/sec
Orbit apsidal rotation capability	Not specified	$\pm$20 deg from hyperbola periapsis
TRAJECTORY PARAMETERS - MARS ORBIT		
Year contamination permitted	2021	1984
Permitted solar occultation by Mars	Maximum 3.7-hr/orbit throughout 6-mo. mission	Not for first 30 days; thereafter limited to 60 min or 8% of orbit period (whichever is smaller)
Permitted Canopus occultation by Mars	Maximum 2-hr / orbit	Not for first 30 days; thereafter limited.
Orbit inclination	Nominal 45 deg	30 to 45 deg
Orbit subperiapsis band	10 deg N to 40 deg S latitude	60 deg S to 40 deg N latitude
Periapsis altitude	1000 km	500 to 1500 km
Apoapsis altitude	20,000 km	10,000 to 20,000 km
Periapsis longitude location	Near the terminator	Near evening or morning terminator
TARGET WEIGHT ALLOCATIONS		
Flt capsule weight	3000 lb	5000 lb
Flt S/C weight (including science)	17,500 lb (excluding adapter)	17,000 lb (including adapter)
S/C science payload weight (including support equipment)	400 lb	400 lb

Table 2-1: COMPARISON OF 1971 AND 1973 INITIAL REQUIREMENTS (Sheet 3 of 4)

REQUIREMENTS	VOYAGER PHASE 1A, TASK B, FINAL REPORT BOEING DOCUMENT D2-82709-6 (1971 Mission)	PERFORMANCE AND DESIGN REQUIREMENTS FOR THE 1973 VOYAGER MISSION, GENERAL SPECIFICATIONS FOR, JAN 1, 1967
GENERAL DESIGN REQUIREMENTS		
Allowable overall probability of contaminating Mars for the Voyager series of missions	1×10^{-4} for any single launch	1×10^{-3} (total of all missions through 1984)
Spacecraft max. dia. Planetary vehicle max, length Spacecraft max. length	240-inches 364-inches 208-inches	240-inches 299-inches 181-inches
Allowable timing error in orbit trim maneuver initiation time	Not specified	< 30 seconds
Maximum allowable unpredictable translational acceleration originating in the P/V	Given as a total distance error of 100 km	0.6×10^{-7} cm/sec^2 (3 σ) time averaged over 1 hour
Maximum alignment error between capsule deorbit thrust axis and primary reference axis	±20 millirad (3 σ)	Not specified
Maximum pointing errors between a preselected inertial direction and S/C primary reference axes at capsule separation	Not specified	±20 millirads (3 σ) ±10 millirads (3 σ)(goal)
Maximum data bit rate to Earth	15,000 bps for a single launch	~8000 bps @ 200×10^6 km per S/C
Storage of science data	Store up to 5×10^7 bits (design goal 5×10^8)	Not specified
Last midcourse maneuver	Not later than 30 days before Mars encounter	Not specified
S/C insertion and capsule deorbit	Insertion in view of DSIF (Goldstone)	Insertion in view of DSIF (Goldstone)

Table 2-1: COMPARISON OF 1971 AND 1973 INITIAL REQUIREMENTS (Sheet 4 of 4)

REQUIREMENTS	VOYAGER PHASE 1A, TASK B, FINAL REPORT BOEING DOCUMENT D2-82709-6 (1971 Mission)	PERFORMANCE AND DESIGN REQUIREMENTS FOR THE 1973 VOYAGER MISSION, GENERAL SPECIFICATIONS FOR, JAN 1, 1967
PROPULSION REQUIREMENTS		
Separation time between P/V arrivals	$\geq$ 10 days	$\geq$ 8 days
ΔV capability for trajectory correction and biasing	200 m/sec per S/C	210 m/sec per S/C
ΔV capability for orbit trim	100 m/sec	150 m/sec
ΔV capability for insertion	$\geq$ 2.0 km/sec 2.2 km/sec (goal)	1.75 km/sec (minimum)
CAPSULE SUPPORT REQUIREMENTS		
Data receipt during descent and entry	50 kbps (maximum rate)	Total of 5×10^6 bits (science)
Capsule separation timing from orbit	3 - 10 days after insertion	3 - 12 days after insertion
Electrical power	200 watts (raw)	200 watts (raw)
DATA RETURN REQUIREMENTS FROM SPACECRAFT TO EARTH		
Interplanetary cruise (science)	2.5×10^5 bits/day	2.5×10^6 bits/day
From orbit (science)	5×10^7 bits/day (end of 6-month orbit life)	6.5×10^8 bits/day (following insertion)
GENERAL CAPSULE REQUIREMENTS		
Capsule separation distance prior to deorbit maneuver	Not specified	300 meters (minimum)
Capsule entry science P/L weight including support equipment	Not specified	45 lb
Capsule surface laboratory science P/L weight including support	Not applicable	130 lb
Maximum timing error in flight capsule release signal	$\pm$1 min (3σ)	$\pm$25 sec (3σ)
Jettison of canisters	10 days prior to Mars encounter	Nonimpact course
Maximum allowable ΔV imparted to spacecraft by capsule release	0.154 m/sec	Not specified

2.2.2 Scope

The requirements were established by using the 1973 Voyager Mars Mission Specification and through the following coordination with MSFC: 1) 1st MSFC/Boeing Technical Coordination Meeting on Task D, 2) MSFC letters R-AS-A67-100 and -121, and 3) MSFC Task D Guidelines dated July 14, 1967. Natural environment requirements were obtained from NASA TMX-5366, while requirements in selected areas were generated through mission and system analyses (Section 3.0).

2.2.3 Results

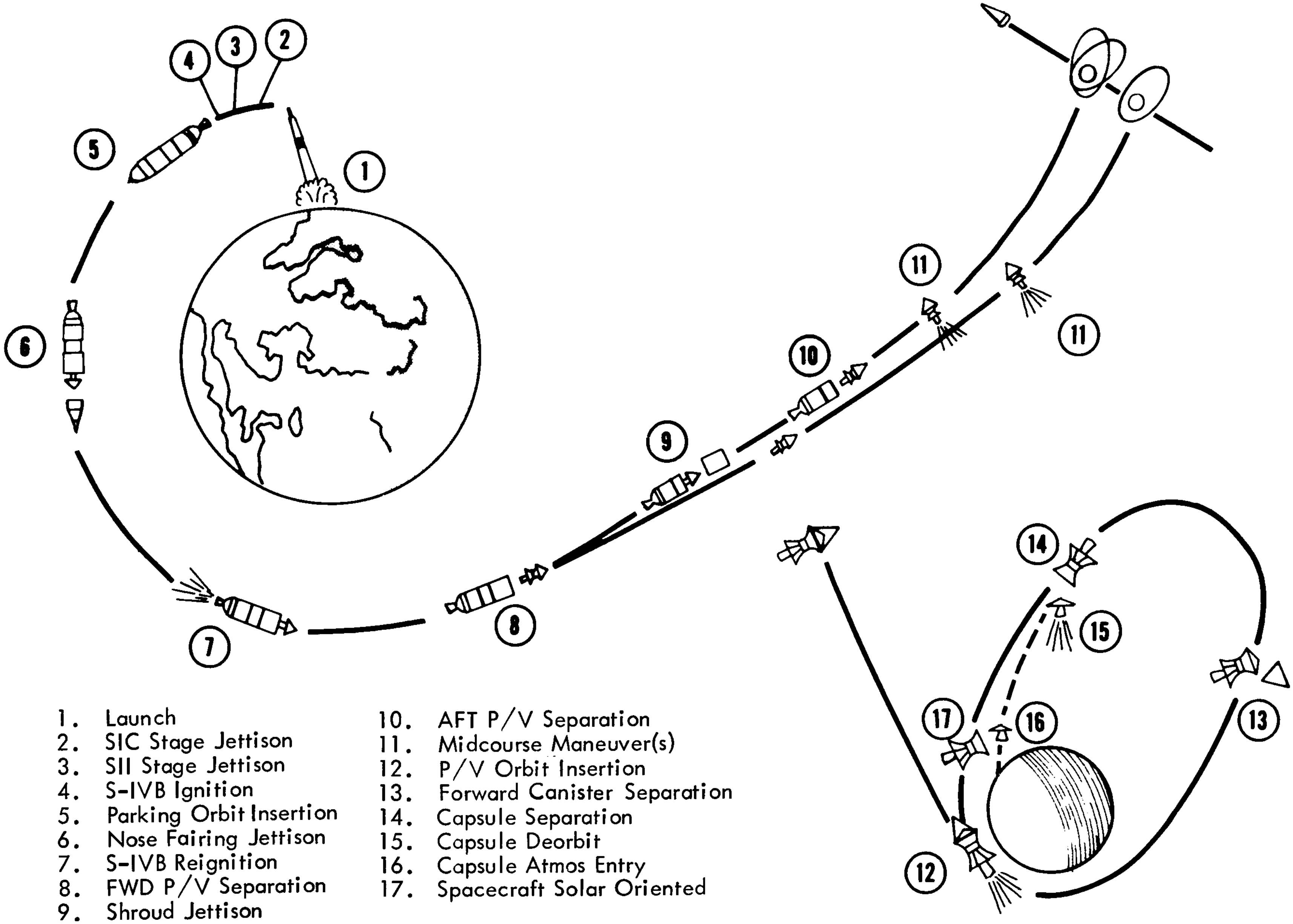
The 1973 requirements are structured into the established subject/categories:

- Mission Definition and Profile--Table 2-2, Figure 2-2
- Mission Requirements--Operational Life--Table 2-3
- Trajectory Parameters--Interplanetary Transfer--Table 2-3, Figures 2-3 through -5
- Trajectory Parameters--Operations Planning--Table 2-4, Figure 2-6
- Trajectory Parameters--Mars Orbit--Table 2-5, Figure 2-7
- Target Weight Allocations--Table 2-6
- General Design Requirements--Tables 2-7 and -8, Figure 2-8
- Propulsion Requirements--Table 2-9
- Capsule Support Requirements--Table 2-10, Figure 2-9
- Data Return Requirements--Table 2-11
- General Capsule Requirements--Table 2-8
- Electrical Power Requirements--Table 2-10
- Reliability Allocations--Tables 2-11 and 2-12
- Spacecraft Stabilization Requirements--Table 2-13
- Science Payload Requirements--Tables 2-13, 2-14 and 2-15

Mission Definition and Profile, Table 2-2, presents an example of how growth potential is built into the flight spacecraft. The propulsion subsystem for the 1973 spacecraft is specified to be compatible with the 1979 mission (maximum weight planetary vehicle) without resizing. Table 2-3 and the associated figures show how versatility is built into the system. This was accomplished by specifying an envelope of performance capability within which the spacecraft must operate, in lieu of a single-point design. To provide a trajectory for spacecraft design and system analysis studies, a specific arrival date was selected for the Mars transfer trajectory (Table 2-4). A baseline Mars orbit for the 1973 spacecraft mission is specified in Table 2-5.

Table 2-2: 1973 REQUIREMENTS

CHARACTERISTICS/ REQUIREMENTS	1973 MISSION
Mission Definition and Profile	Two P/V's shall be inserted into orbits around Mars. The flight capsule of each P/V shall be separated in Mars orbit and shall descend to the surface and land. After landing, the surface laboratory aboard the flight capsule shall operate on the surface and the flight S/C shall operate in orbit for specified periods of time. The boost shroud separation shall be over the nose with the forward shroud section ejected in Earth orbit. The two P/V's shall be identical. Minor differences in the spacecraft science subsystem will be considered The 1973 Voyager mission shall be designed using a nominal capsule weight of 6000 lb. The 1973 propulsion module shall be sized for the 1979 mission which includes a 7000 lb Flight Capsule. A 5000 lb project contingency shall be used in the launch vehicle payload allocation.

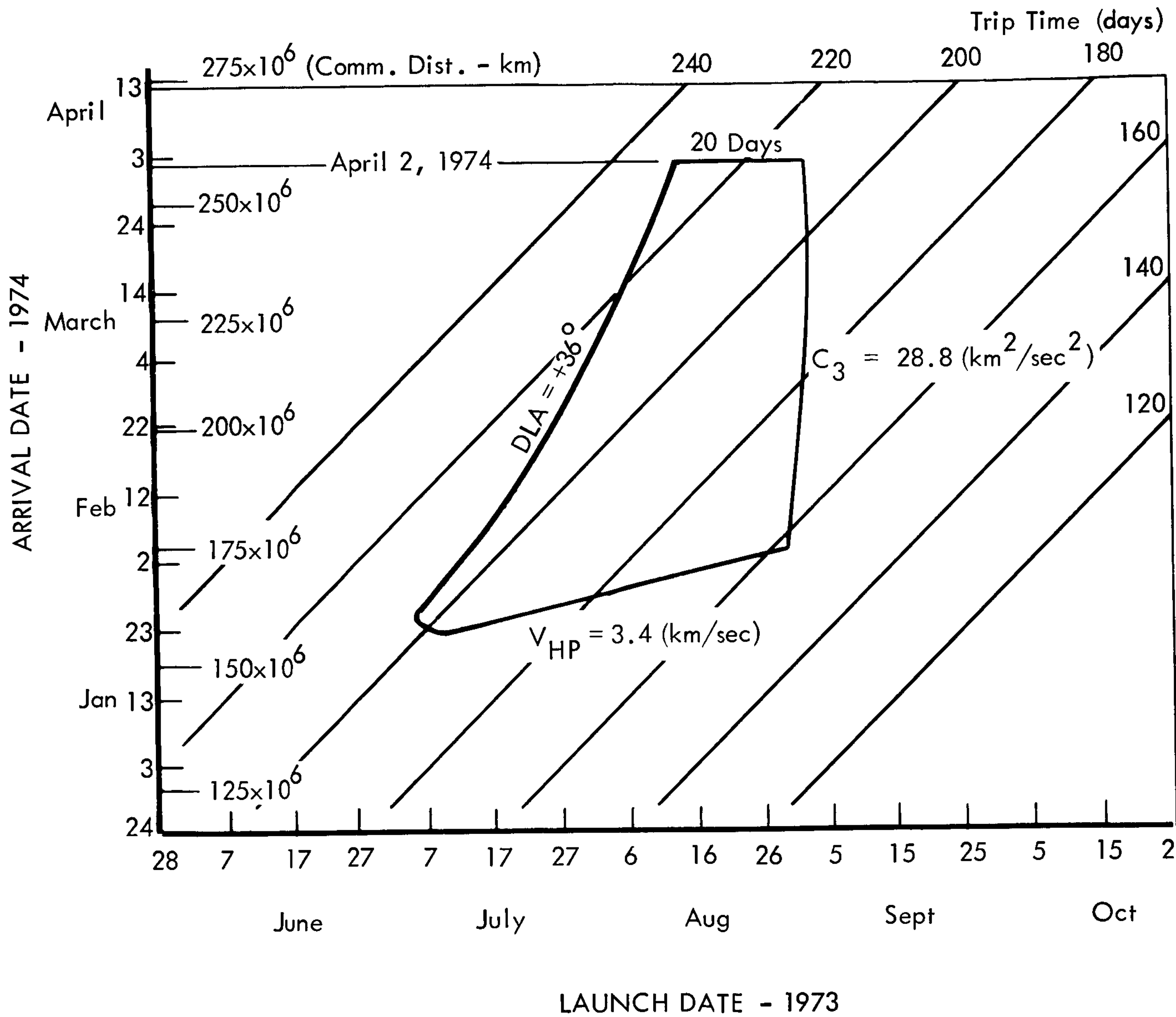


1. Launch
2. SIC Stage Jettison
3. SII Stage Jettison
4. S-IVB Ignition
5. Parking Orbit Insertion
6. Nose Fairing Jettison
7. S-IVB Reignition
8. FWD P/V Separation
9. Shroud Jettison
10. AFT P/V Separation
11. Midcourse Maneuver(s)
12. P/V Orbit Insertion
13. Forward Canister Separation
14. Capsule Separation
15. Capsule Deorbit
16. Capsule Atmos Entry
17. Spacecraft Solar Oriented

Figure 2-2: VOYAGER MISSION PROFILE

Table 2-3: 1973 REQUIREMENTS

CHARACTERISTICS/ REQUIREMENTS	1973 MISSION
Mission Requirements - Operational Life	Single Launch, two P/V's. Six months life in orbit. Two days life on surface. Transmit science and engineering data to Earth. Minimum launch period of 20 days.
Trajectory Parameters - Interplanetary Transfer	Maximum ranges of pertinent parameters are: • Attainable launch period (days) for fixed arrival date - 50 • Shortest communication distance at encounter (km x 10^6) - 158 • Longest communication distance at encounter (km x 10^6) - 260 • Maximum communication distance during 6 months in orbit considering latest arrival date (km $\times$ 10^6) - 393 (Greatest distance in the 70's) • Earliest arrival date - 1/23/74: Latest arrival date - 4/2/74 • Shortest trip time (days) - 157 • Longest trip time (days) - 231 • Shortest Sun-Mars distance at arrival (AU) - 1.573 • Longest Sun-Mars distance at arrival (AU) - 1.640 • Longest Sun-Mars distance during 6 months in orbit considering latest arrival date (AU) - 1.670



BASIS:

- Type I trajectories
- C_3 limit based on weight allocations for 6,000 lb capsule (case B).
- V_{HP} limit based on 500 km by 18,500 km 12.4 hr. orbit, with $\pm$ 20 degree apsidal rotation.

Figure 2-3: TRAJECTORY PARAMETERS — INTERPLANETARY TRANSFER

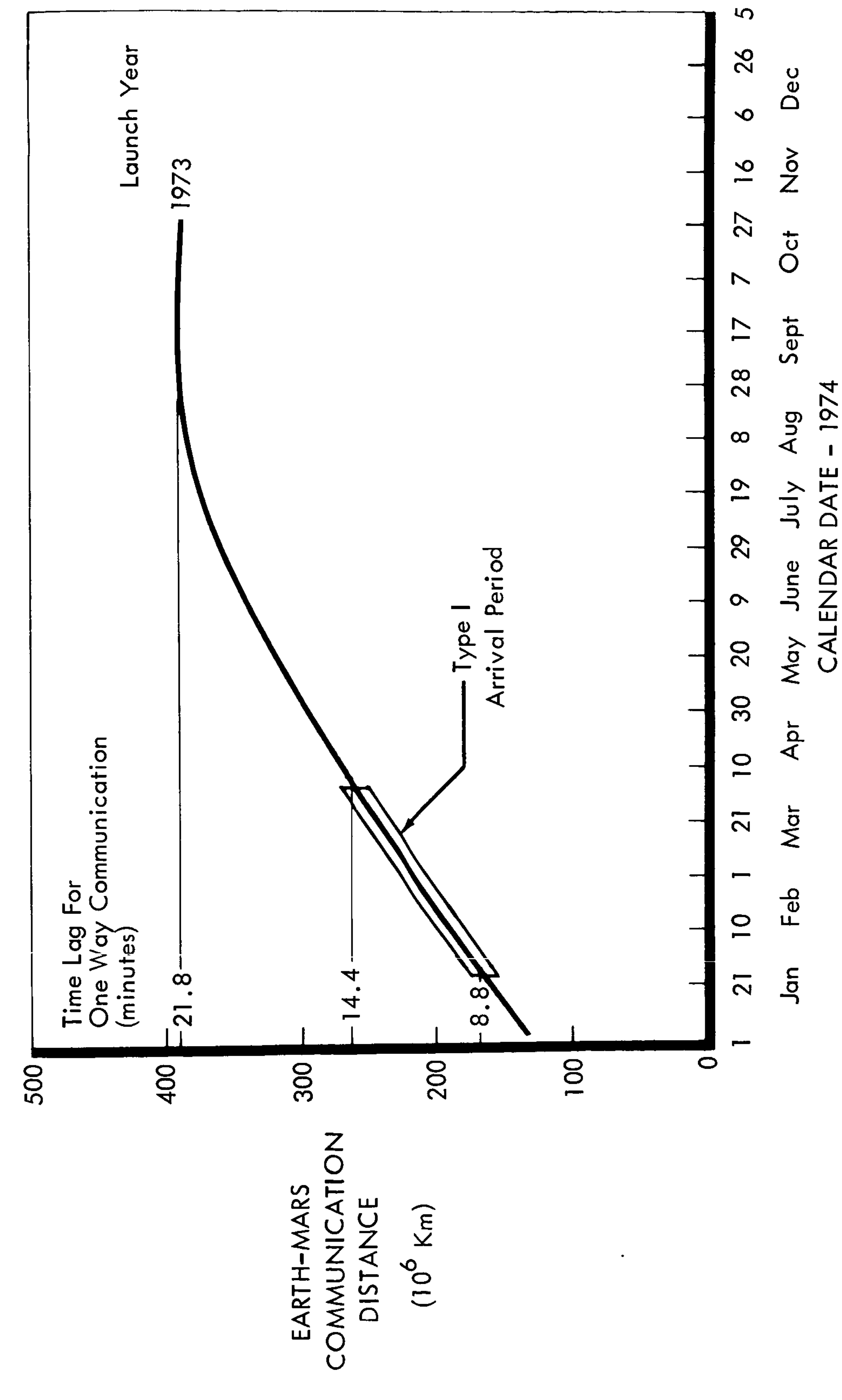


Figure 2-4: EARTH-MARS SEPARATION RANGE

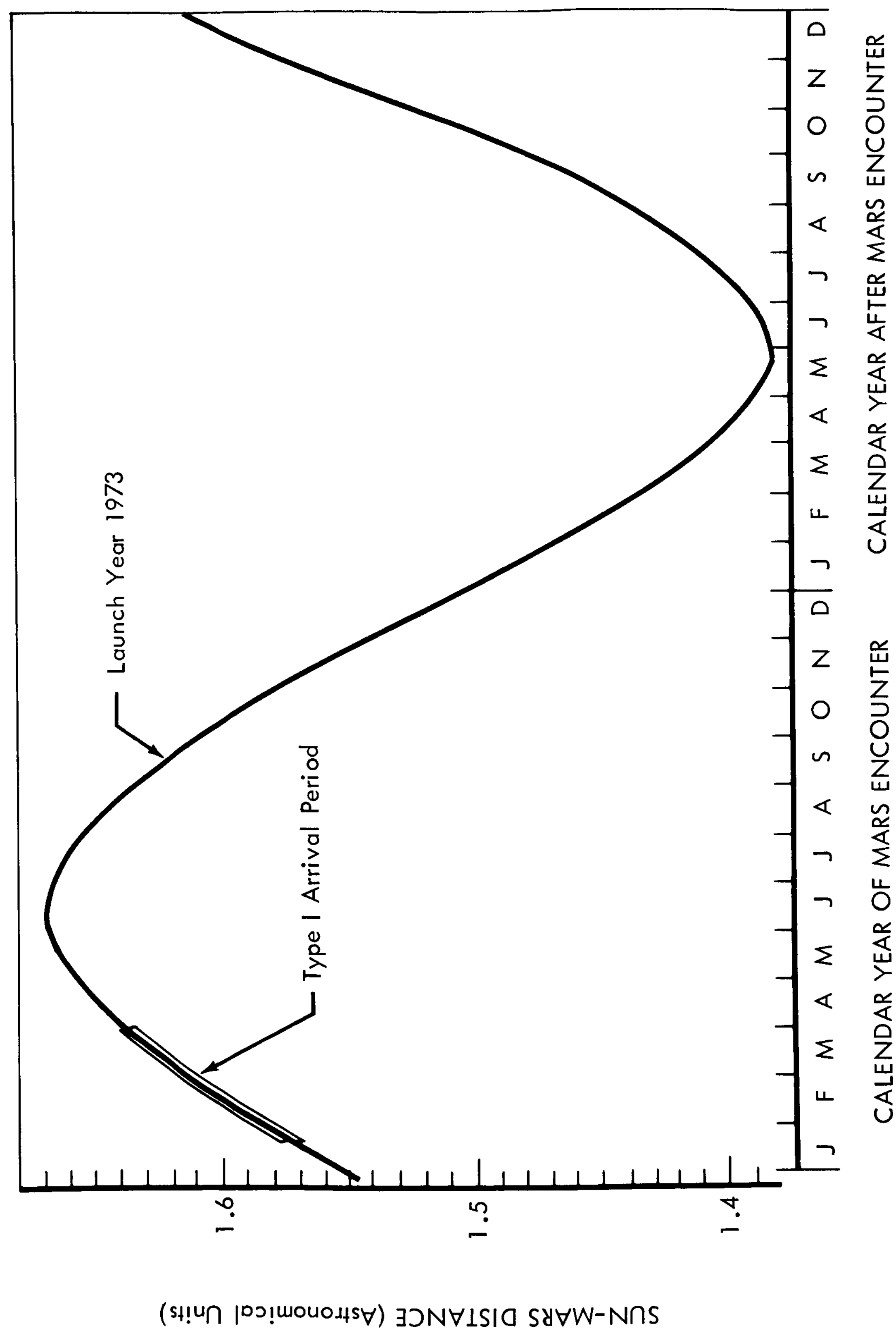


Figure 2-5: SUN-MARS SEPARATION RANGE

Table 2-4: 1973 REQUIREMENTS

CHARACTERISTICS/ REQUIREMENTS	1973 MISSION	
Trajectory Parameters - Operations Planning	Trajectory type	1
	Launch period (days)*	20
	C_3 range (km^2/sec^2)**	19.1/28.8
	Flight time range (days)*	212/231
	Communication distance at arrival (km x 10^6)	260
	Sun-Mars distance at arrival (AU)	1.64
	V_{HP} range (km/sec)*	2.48/2.55
	Launch azimuth (deg East of North)	90-115
	Launch date	8/14 - 9/3/73
	Arrival date	4-2-74
	Apsidal rotation will be required at orbit insertion to place the periapsis at a favorable Sun location	
	The aiming point ($\bar{B}$) shall be biased to satisfy a 0.2×10^{-5} probability allocation for accidental impact at encounter for each planetary vehicle.	
	* Trajectory range is designed to give a constant arrival date at Mars. ** A 6,000 lb capsule is assumed.	

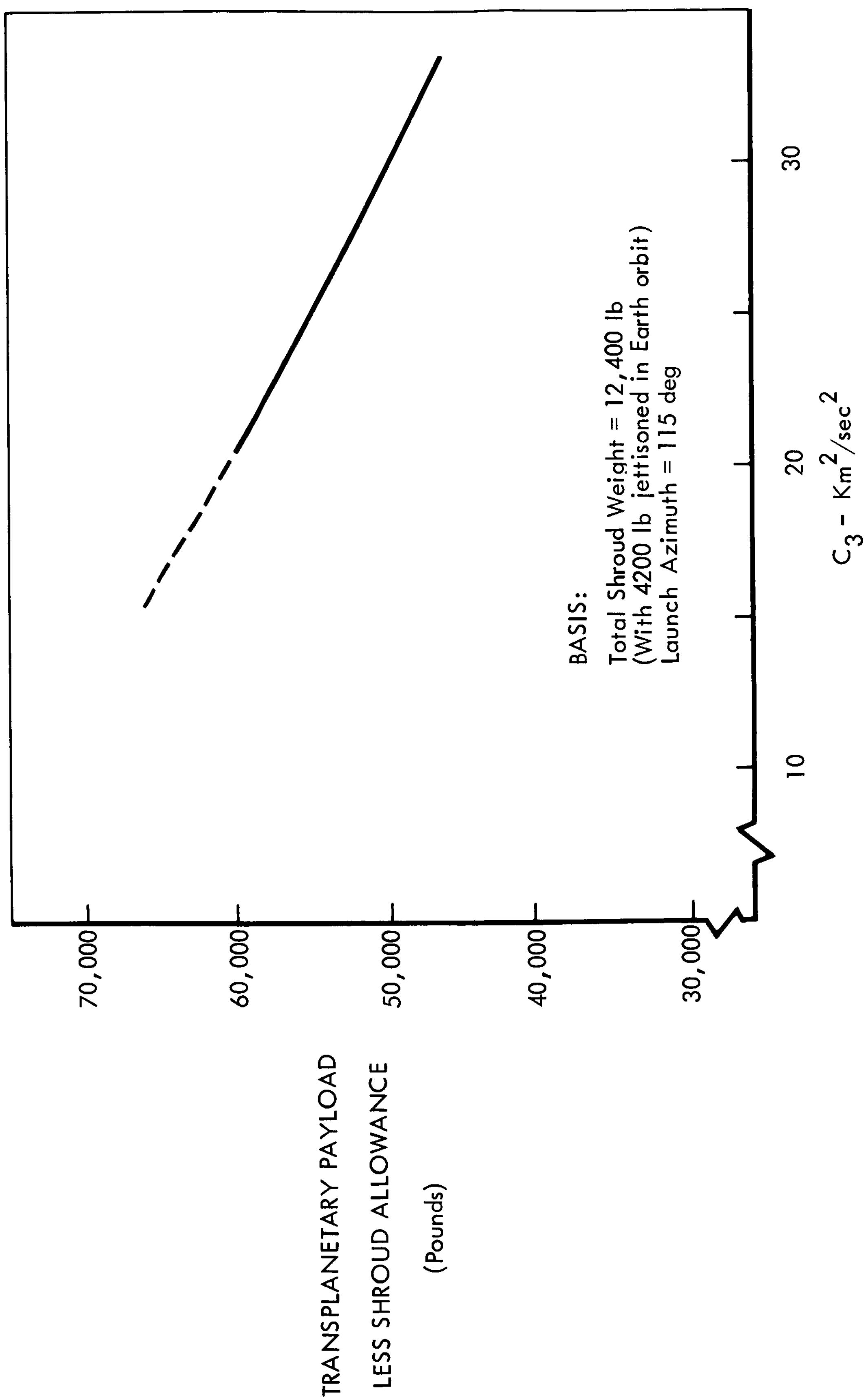


Figure 2-6: C_3 VS SATURN V PAYLOAD CAPABILITY

Table 2-5: 1973 REQUIREMENTS

CHARACTERISTICS/ REQUIREMENTS	1973 MISSION
Trajectory Parameters - Mars Orbit	Orbit shall not give solar accultation for first 30 days, thereafter not to exceed the lesser of 1.5 hours or 12.5% of the orbit period. Periapsis altitude - 500 km Apoapsis altitude - 18,500 km Orbit inclination - 45 degrees Subperiapsis point - within latitude band from 60 degrees S to 40 degrees N latitude Illumination angle at periapsis - 15 degrees Periapsis location - near evening terminator Orbit period - 12.4 hours Orbit insertion to be in view of Goldstone, California Orbit lifetime - 11 years

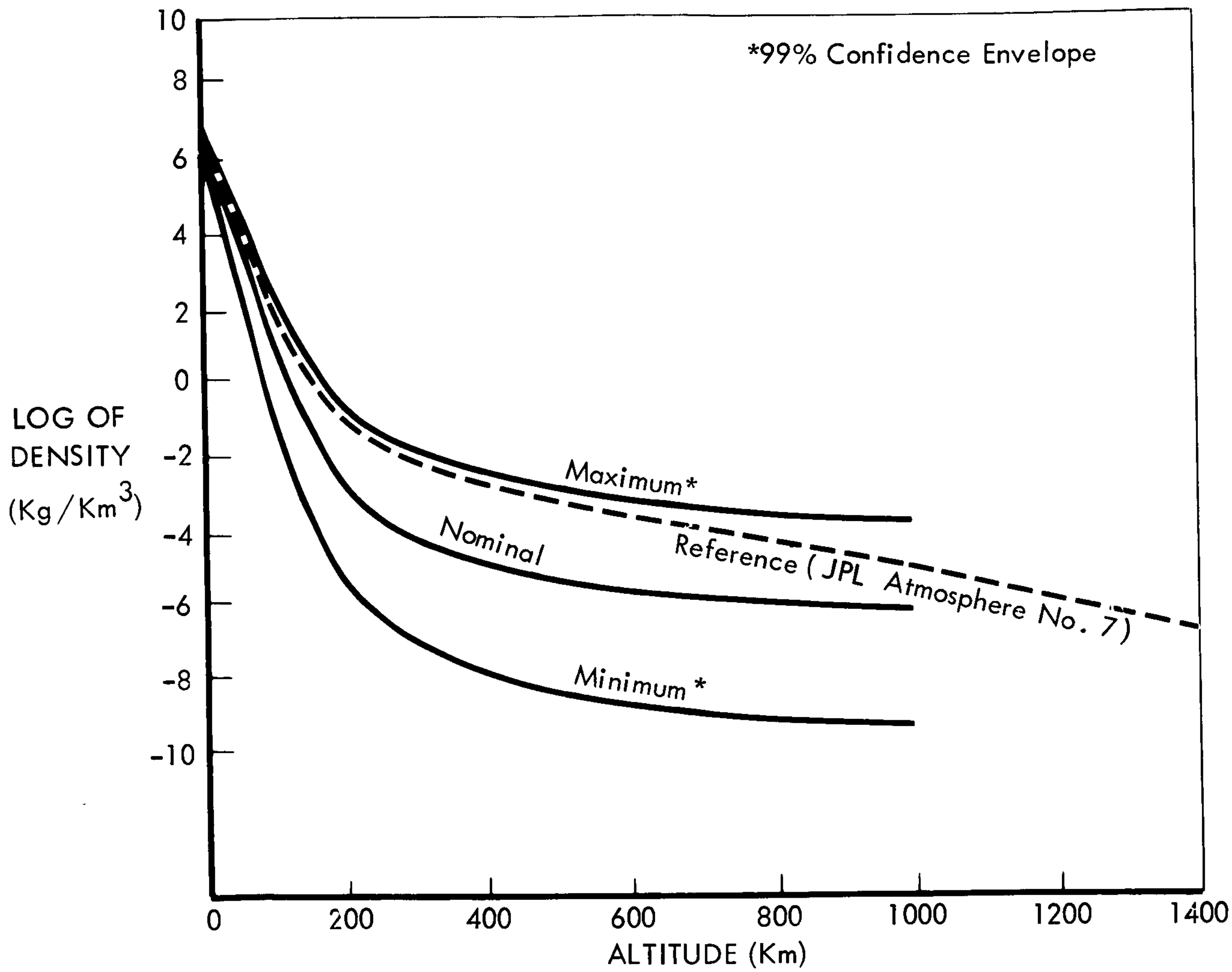


Figure 2-7: MARS ATMOSPHERE DENSITY MODEL

Table 2-6: 1973 REQUIREMENTS

CHARACTERISTICS/ REQUIREMENTS	1973 MISSION				
Target Weight Allocations			Case A		Case B
	Total Dry Flight Spacecraft		5,650 lb		5,800 lb
	Propulsion system inerts (including residuals)	2050		2100	
	S/C bus structure & equipment *	2940		3030	
	Spacecraft science **	390		390	
	Weight contingency (5% of dry weight)	270		280	
	Usable Propellants		10,100 lb		10,950 lb
	Flight Capsule ***		5,000 lb		6,000 lb
	P/V Adapter ****		250 lb		250 lb
	Total Planetary Vehicle Plus Adapter		21,000 lb		23,000 lb

* Includes 50 lb of capsule bus support equipment.
** Includes instruments and DAE only. Scan platform(s) structure and mechanisms included in the spacecraft bus weight allocation.
*** Includes sterilization canister of which 650 lb may remain with spacecraft after capsule separation.
**** Does not remain with spacecraft.

Table 2-7: 1973 REQUIREMENTS

CHARACTERISTICS / REQUIREMENTS	1973 MISSION
General Design Requirements	Support the flight capsule, propulsion module, and science payload, as well as supporting the subsystem components and providing protection from meteoroids. The solar array shall be a fixed static design with provisions to add deployable panels. The P/V arrangement and dynamic envelope shown in Figure 2-8 shall be used at least for the initial design. The adapter which mates the planetary vehicle to the load carrying boost shroud can be a separate piece of hardware or an "integral adapter" that is part of the flight spacecraft basic design. Withstand 1) the natural environment as stated in NASA TMX-53616 and 2) the induced environment.

Table 2-8: 1973 REQUIREMENTS

CHARACTERISTICS/ REQUIREMENTS	1973 MISSION
General Design Requirements (continued)	There are no requirements for decontamination or sterilization of any part of the flight spacecraft. To keep microbial loading down, the flight spacecraft shall be assembled in a Class 100,000 clean room per Federal Standard No. 209a dated August 10, 1966. Gaseous propellants and pressurants (N_2 and He) shall be filtered and loaded ascepticaly. Efforts shall be made to use nonmagnetic materials and to keep the magnetic field source low if costs aren't shown to be prohibitive. The Canopus sensor shall be located such that it has an unobstructed field of view of at least $\pm$ 45 degrees at all times with respect to surrounding S/C structure and components. For cost effectiveness, off-the-shelf components and subsystems will be nonstandard packaging design; standardized packaging design will be considered for "make" equipments and new "buy" equipments. The propulsion module shall be sized for the 1979 mission, which includes a 7000-lb flight capsule. Provisions shall be made for a destruct system in accordance with range safety requirements. Destruct shall be accomplished by rupturing the propellant tanks in each spacecraft and capsule. It shall be assumed that the ordnance are attached to the shroud structure and are jettisoned along with the shroud.
General Capsule Requirements	The lower portion of the capsule sterilization canister will remain with the spacecraft following capsule jettison (up to 650 lb). A maximum outside diameter of 20 feet shall be assumed for the flight capsule.

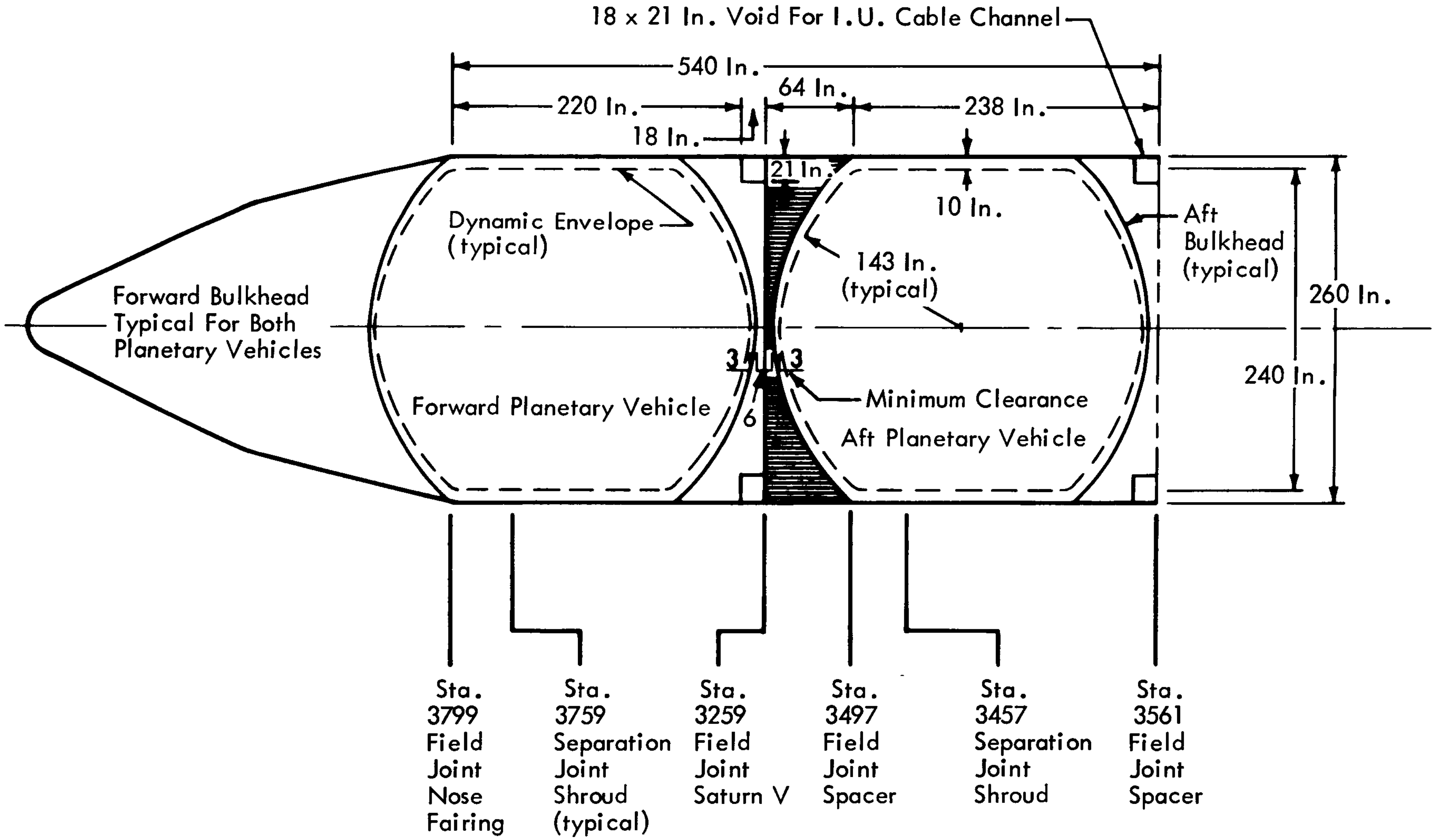


Figure 2-8: SATURN V VOYAGER PLANETARY VEHICLE DYNAMIC ENVELOPE

Table 2-9: 1973 REQUIREMENTS

CHARACTERISTICS / REQUIREMENTS	1973 MISSION
Propulsion Requirements	Provide velocity increments for: Two midcourse corrections One arrival biasing maneuver (8 days) including a correction maneuver Orbit insertion Two orbit trim maneuvers The propulsion system shall utilize: Modified LMDE for orbit insertion Engine gimballing for pitch - yaw thrust vector control Reaction control jets for roll control Minimum velocity requirement allotments shall be: Trajectory corrections plus arrival biasing (m/sec) 210 Orbit insertion (m/sec) 1590 Orbit trim (m/sec) 150 Total 1950 The propulsion module shall be a single point design capable of performing Mars missions through the 1979 opportunity. Total ΔV is the same for 1973 - 1979 missions. Consideration shall be given to the Agena engine (Model 8517) and the Titan transtage engine as modular replacement engines for the LMDE.

Table 2-10: 1973 REQUIREMENTS

CHARACTERISTICS/ REQUIREMENTS	1973 MISSION
Capsule Support Requirements	Deliver and release capsule from Mars orbit. Receive capsule data (50-200 Kbps) during the entry phase and provide a communication relay mode for transmittal of data from the surface. Supply 200 watts of raw power to capsule and 50 watts (receiver power) to S/C mounted capsule equipment. Relay commands from Earth to capsule prior to separation. Observe capsule from orbit. Shield capsule from Sun when the P/V is on its on-Sun operational mode. The capsule bus support equipment aboard the S/C shall be assumed to weigh 50 lb.
Electrical Power Requirements	Provide power to the flight capsule and capsule bus support equipment (see capsule support requirements). Provide and distribute power to all component consumers aboard the flight spacecraft throughout the mission life. The average power load allocation for each operational phase shall be assumed as shown in Figure 2-9.

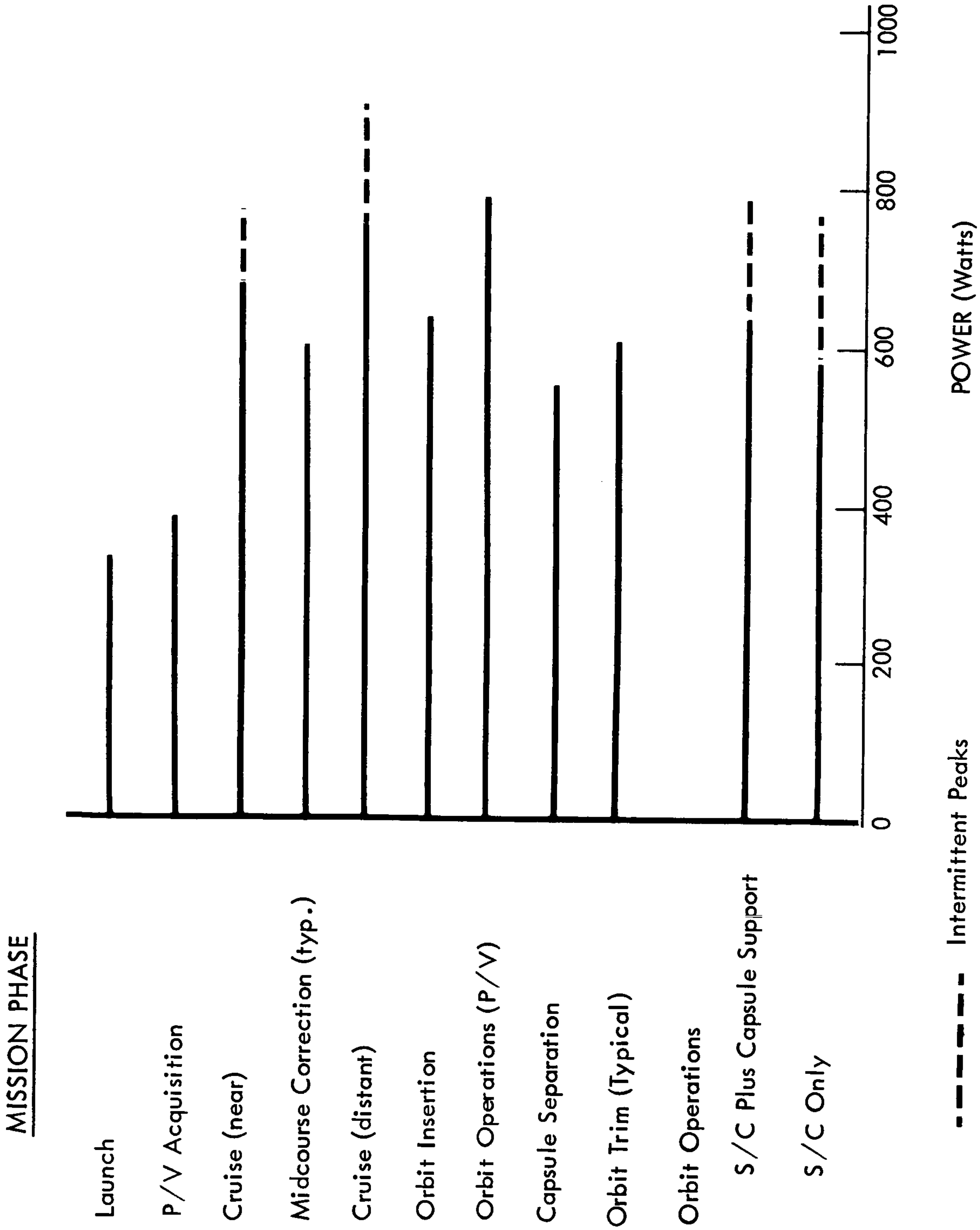


Figure 2-9: AVERAGE POWER ALLOCATIONS — 1973 Mission

Table 2-11: 1973 REQUIREMENTS

CHARACTERISTICS/ REQUIREMENTS	1973 MISSION
Data Return Requirements	Communications shall be required between the flight spacecraft and Earth stations for all flight phases: • Launch to heliocentric trajectory • Cruise • Midcourse corrections • Orbit insertion • Orbit operations • Capsule entry • Landed operations The average data bit rate requirements for orbit operation shall be 12,500 bps. The spacecraft shall also provide the capability of receiving data from the flight capsule as follows: • A total of 5×10^6 bits (science) during orbital descent and entry through the atmosphere. • A total of 5×10^6 bits of science data per day after landing
Reliability Allocations	Numerical reliability allocations are shown in Table 2-12.

Table 2-12: RELIABILITY ALLOCATIONS — 1973 Mission

SYSTEM ELEMENTS		ALLOCATION PRIME	SUB	SUB
Flight Spacecraft		0.747		
Spacecraft Bus			0.750	
Power Subsystem				0.991
Computing & Sequencing Subsystem				0.942
Guidance & Control Subsystem				0.983
Radio Subsystem				0.962
Antenna Subsystem				0.995
Telemetry Subsystem				0.995
Data Storage Subsystem				0.923
Structual & Mechanical Subsystem				0.936
Pyrotechnic Subsystem				0.9999
Temperature Control Subsystem				0.9999
Cabling Subsystem				0.9940
Propulsion Subsystem			0.995	
Total Spacecraft (Excluding Science Payload)	1 of 2	0.936		
Science Payload (Estimated)			*0.322	
Total Spacecraft (Including Science Payload)	Each	0.240		
	1 of 2	0.422		

*Assessment value used pending a total system trade study.

Table 2-13: 1973 REQUIREMENTS

CHARACTERISTICS / REQUIREMENTS	1973 MISSION
Spacecraft Stabilization Requirements	Provide three-axis control at all times. During periods when the P/V is oriented along the Sun line, attitude position signals shall be provided by Sun and Canopus sensors. A strap-down inertial reference unit shall provide position data during maneuvers and periods of Sun and Canopus occultations. Attitude positioning is provided by reaction control.
Science Payload Requirements	Spacecraft science data will be taken during the following periods: • Trans-Mars cruise (interplanetary measurements) • Mars approach (interplanetary measurements plus photo-imaging of planet) • Mars orbit (all instruments in operation) The science payload is comprised of instruments, data automation equipment, as well as supporting hardware and electronic parts. The payload definition is shown for the initial design definition task in terms of weight, Table 2-14, and power, Table 2-15.

Table 2-14: 1973 VOYAGER MARS SCIENCE PAYLOAD ALLOCATION

ITEM	WEIGHT		
	Platform (lb)	Body (lb)	Total (lb)
● Scan Platform Instruments (Orbital)			
Platform 1			
Photoimaging television (3)	150		150
Mars IR radiometer	20		20
IR spectrometer, high resolution	30		30
IR spectrometer, broad band	25		25
Atmosphere Polarimeter	9		9
Atmosphere Mass Spectrometer	8		8
Platform 2			
UV spectrometer	30		30
● Body Mounted Instruments (Cruise)			
Plasma probe		10	10
Cosmic ray telescope		8	7
Cosmic dust detector		5	5
Trapped radiation detector		3	3
Ion chamber (boom mounted)		3	3
Total Instruments	272	29	300
Scan Platform No.1			--*
Platform No.2			--*
DAE			50
Harness & Miscellaneous (including power switching electronics)			20
Contingency			20
Total Weight Allocation			390

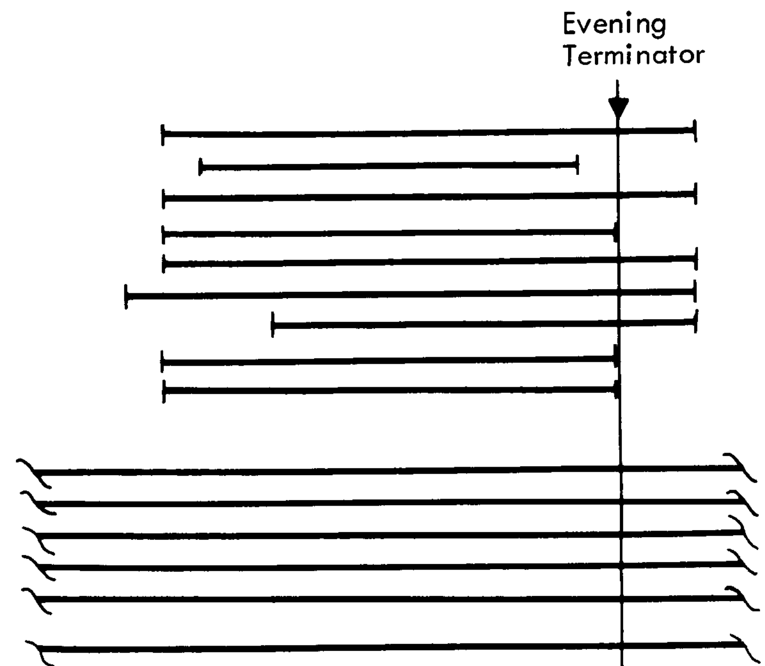
* Weight incorporated in the spacecraft bus allotment and Equipment Allocation

● POWER REQUIREMENTS

Payload Element	Power (watts)	On Time (min.)
Platform No. 1	10*	69
Photo Imaging TV (3)	60	49
Mars IR Radiometer	6	69
IR Spectrometer, HI-Res.	14	59
IR Spectrometer, Broad Band	5	69
Atmosphere Polarimeter	3	75
Atmos. Mass Spectrometer	10	55
Platform No. 2	5*	59
UV Spectrometer	12	59
Body Mounted		
Plasma Probe	3	Cont.
Cosmic Ray Telescope	3	Cont.
Cosmic Dust Detector	1	Cont.
Trapped Radiation Detector	1	Cont.
Ion Chamber	1	Cont.
DAE and Power Switching	29	Cont.
Total Power	163 watts	

*Average power considering both on and off times

● POWER ON-OFF CYCLE



● POWER PROFILE

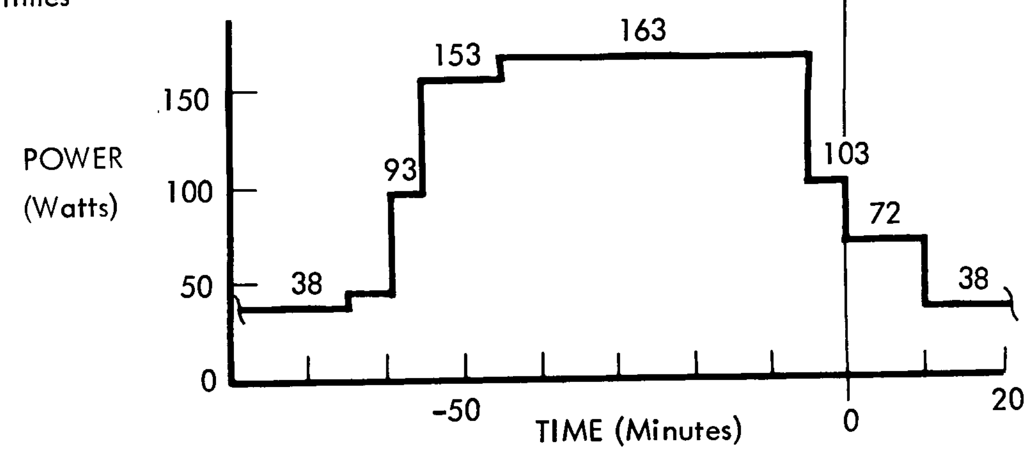


Table 2-15: SCIENCE PAYLOAD POWER PROFILE DURING ORBIT

Within the mission requirements expressed above, an iterative evaluation of spacecraft parameters was conducted to develop a practical and feasible set of spacecraft requirements. As a result, the target weight allocations specified in Table 2-6 were established. Two weight statement allocations are provided to accommodate two potential capsule weights for the 1973 mission.

The planetary vehicle dynamic envelope, shown in Figure 2-8, in conjunction with data in Tables 2-7 and 2-8, control the general design features of the planetary vehicle.

The developed mission and system requirements and significant subsystem parameters, (e.g., propellant loading, power, quantity of data) interact to affect the spacecraft design. An iterative process of requirement versus implementation was conducted at the system level. Specific design areas analyzed in this manner are:

1) Propulsion subsystem--Table 2-9

2) Capsule support and electrical power--Table 2-10

3) Data allocation and transmission--Table 2-11

4) Spacecraft stabilization--Table 2-13.

Reliability allocations, Table 2-12, were developed for the spacecraft and its subsystems. Previous analyses (Phase 1A, Task B data, and Voyager equipment definition) were extended to cover the 1973 mission profile. The science subsystems allocation was established by assessing the hypothetical 1973 science payload (Table 2-14) as a single thread subsystem (every experiment must work). This approach establishes meaningful reliability allocations, but cannot be used for inferring probability of mission success. Mission success must be based on a criterion that accounts for the relative value of each experiment through the life of the mission. Repetitive data is not necessarily valuable, nor are all experiments of equal value in fulfilling the missions scientific objectives. A criterion might be established in which the probability of mission success approaches the reliability allocation of 0.746 for a single flight spacecraft excluding the science payload. If the mission success criteria required return of all the science data for an entire 6-month initial mission, the reliability allocation of 0.24 for a single flight spacecraft including the science payload is synonomous with the probability of mission success of a single spacecraft.

Science payload requirements for 1973, Tables 2-13, 2-14, and 2-15, were developed to identify the impact of the experiments payload on spacecraft performance such as power, weight, stability, and pointing capability. These requirements were developed (see D2-115002-4, Section 4.0) by taking into consideration three aspects of the science subsystem: (1) usefulness in supporting 1973 mission objectives, 2) compatibility with the spacecraft and its mission profile, and 3) logical evolution of the spacecraft science payload through the first four Mars-Voyager missions (1973, 1975, 1977, 1979).

2.3 PRELIMINARY 1975-1977-1979 SPACECRAFT REQUIREMENTS

2.3.1 Purpose

The purpose was to establish preliminary spacecraft requirements for 1975-1977-1979 Mars Voyager missions.

2.3.2 Scope

The science payload and flight capsule evolution of the 1975-1977-1979 Voyager missions leads to new requirements. The unique 1975-1977-1979 mission and system requirements were developed on the basis of new science and flight capsule payloads and the different Earth-Mars trajectory relationships for these subsequent missions.

2.3.3 Results

Table 2-16 summarizes the changes within the 15 requirement subject headings that are affected by the 1975, 1977, and 1979 missions. Because many of these effects were reflected in the 1973 requirements, the impact of requirement changes on selected areas of spacecraft design is minimized. A major consideration of the 1977 and 1979 missions is the increase in orbital life from 6 to 12 months (Table 2-16).

The trajectory envelopes for Type I transfers in the 1973 to 1979 missions, and Type II transfers in 1975 and 1977, are shown in Figure 2-10. Trajectory parameters are summarized in Tables 2-17 and 2-18.

The trajectory envelopes are consistent in the following details:

1) Total propulsive velocity change (ΔV) is 1950 m/sec. The minimum required orbit insertion ΔV is defined for each year by subtracting the ΔV allocations for (a) midcourse trajectory correction, (b) arrival time biasing, and (c) orbit trim maneuvers for that mission from the total ΔV.

2) The conversion from allocated ΔV to the hyperbolic excess velocity (V_{HP}) limits, required for trajectory envelope definition, is based on a nominal 500-km periapsis, 12.4-hour orbit with assumed apsidal rotations of $\pm$ 20 degrees. The changes in minimum required orbit insertion ΔV for the four missions reflect the changing weight allocations and result in different V_{HP} limits.

3) Launch vehicle capability is computed for each launch opportunity by consideration of:

- Target weight allocations, Tables 2-19 and 2-20.
- A constant 5000-pound project weight contingency for each mission.
- Geocentric launch energy (C_3) versus payload for the AS 506 Saturn V launch vehicle including shroud allowance considerations.

Table 2-16: 1975-1977-1979 PERFORMANCE AND DESIGN REQUIREMENTS (Sheet 1 of 2)

CHARACTERISTICS/REQUIREMENTS	CHANGES* 1975	1977	1979
Mission Definition and Profile			
Nominal flight capsule weight (lb)	7000	7000	7000
Alternate flight capsule weight (lb)	5000	5000	5000
Mission Requirements - Operational Lifetime			
Spacecraft lifetime in orbit (months)	6	12	12
Lander lifetime on surface (months)	6	12	12
Trajectory Parameters - Interplanetary Transfer			
Maximum ranges of pertinent parameters	Table 2-17		
Trajectory design envelopes	Figure 2-10		
Trajectory Parameters - Operations Planning	None		
Trajectory Parameters - Mars Orbit	Table 2-18		
Target Weight Allocations			
For design condition flight capsules	Table 2-19		
For alternate flight capsule	Table 2-20		
General Design Requirements	Accommodate capsule RTG		

* Changes from 1973 performance and design requirements, Section 2.2.

Table 2-16: 1975-1977-1979 PERFORMANCE AND DESIGN REQUIREMENTS (Sheet 2 of 2)

CHARACTERISTICS/REQUIREMENTS	CHANGES* 1975	CHANGES* 1977	CHANGES* 1979
Propulsion Requirements			
Total velocity allotment (m/sec) - minimum	1950	1950	1950
Trajectory correction plus arrival biasing (m/sec)	75	75	110
Orbit insertion (m/sec)	1775	1775	1740
Orbit trim (m/sec)	100	100	100
Capsule Support Requirement	No power to capsule		
Data Return Requirements			
Data received from capsule (bits/day)	1×10^7		
Earth-Mars communication distance	Figure 2-11		
General Capsule Requirements	No changes		
Electrical Power Requirements			
Solar intensity	Figure 2-12		
Average power allocation for operational phases	Table 2-21		
Reliability Allocations	None		
Spacecraft Stabilization Requirements	No Change	Table 2-22	
Science Payload Requirements			
Instrument listing - weights	Table 2-23		
Orbital power profile	Figure 2-13	Figure 2-14	Figure 2-15

* Changes from 1973 performance and design requirements, Section 2.2.

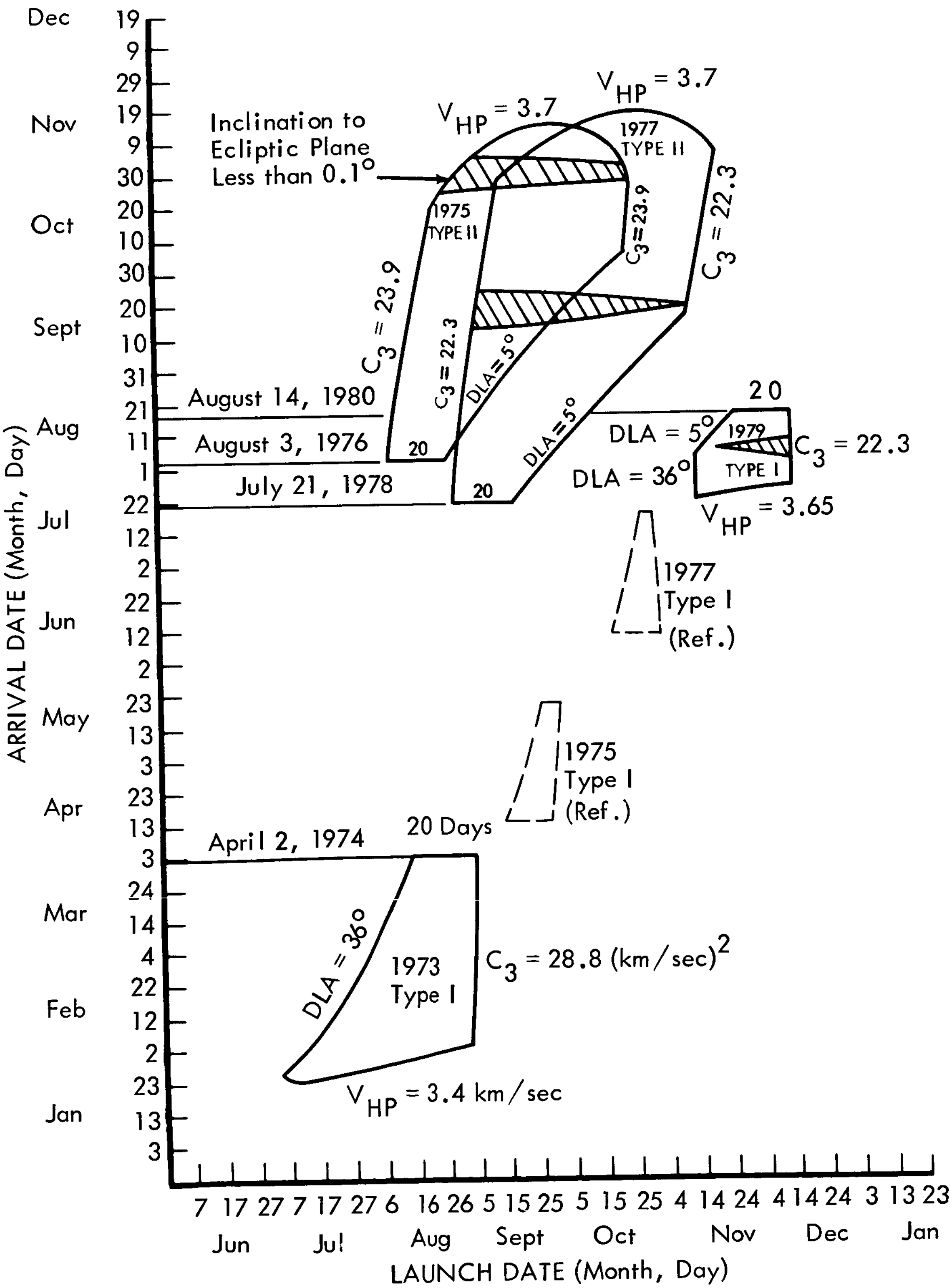


Figure 2-10: TRAJECTORY DESIGN ENVELOPES

Table 2-17: TRAJECTORY PARAMETER RANGES

	MISSION YEAR			
	1973	1975	1977	1979
Trajectory type	I	II	II	I
Attainable launch period (days) for fixed arrival date	50	63	66	30
Shortest communication distance at encounter (Km x 10^6)	158	352	300	242
Longest communication distance at encounter (Km x 10^6)	260	380	365	268
Maximum communication distance during orbit (Km x 10^6)	393	380	365	360
Earliest arrival date	1/23	8/3	7/21	7/18
Latest arrival date	4/2	11/12	11/17	8/14
Shortest trip time (days)	157	342	306	222
Longest trip time (days)	231	422	408	266
Shortest Sun-Mars distance at arrival (AU))	1.573	1.525	1.465	1.575
Longest Sun-Mars distance at arrival (AU))	1.640	1.630	1.610	1.535
Longest possible Sun-Mars distance (AU)	1.670	1.630	1.610	1.575
Shortest possible Sun-Mars distance (AU)	1.573	1.380	1.380	1.380

Table 2-18: MARS ORBITAL PARAMETERS FOR PRELIMINARY SYSTEM DESIGN

	MISSION YEAR			
	1973	1975	1977	1979
Periapsis Altitude (km)	500			
Apoapsis Altitude (km)	18,500			
Inclination (degrees)	45			
Periapsis Location				
Latitude	60S-40N	10N - 15N		
Illumination Angle (degrees)	15° - Near evening terminator			
Orbit Period (hours)	12.4			
Orbit Insertion Constraint	In view of Goldstone, California			
Minimum Orbit Lifetime (years)	11	9	7	5

Table 2-19: TARGET WEIGHT ALLOCATIONS (Pounds)

	MISSION YEAR			
	1973	1975	1977	1979
Total Planetary Vehicle	22,750	25,350	26,250	26,250
Total Dry Flight Spacecraft	5,800	6,150	6,600	6,600
Propulsion system inerts	(2,100)	(2,100)	(2,100)	(2,100)
Spacecraft bus and equipment*	(3,030)	(3,110)	(3,200)	(3,200)
Orbital science**	(390)	(650)	(1,000)****	(1,000)****
Contingency (5% dry weight)	(280)	(290)	(300)	(300)
Usable Propellants	10,950	12,200	12,650	12,650
Flight Capsule***	6,000	7,000	7,000	7,000
Planetary Vehicle Adapter Allowance	250	250	250	250
Boosted weight per planetary vehicle	23,000 lb	25,600 lb	26,500 lb	26,500 lb

* Includes 50-lb capsule bus support equipment plus 120-lb planet scan platform and science booms in 1973 and 1975.
** Includes instruments and DAE only.
*** Includes sterilization canister of which 650 lb may remain with the spacecraft.
**** 650-lb fixed mount camera system replaces scan platform.

Table 2-20: TARGET WEIGHT ALLOCATIONS — Alternate Flight Capsule Weights

	MISSION YEAR			
	1973	1975	1977	1979
Total Planetary Vehicle	20,750	21,450	22,250	22,250
Total Dry Flight Spacecraft	5,650	6,000	6,450	6,450
Propulsion system inerts	(2,050)	(2,050)	(2,050)	(2,050)
Spacecraft bus and equipment*	(2,940)	(3,020)	(3,110)	(3,110)
Orbital science**	(390)	(650)	(1,000)****	(1,000)****
Contingency (5% dry weight)	(270)	(280)	(290)	(290)
Usable Propellants	10,100	10,450	10,800	10,800
Flight Capsule***	5,000	5,000	5,000	5,000
Planetary Vehicle Adapter Allowance	250	250	250	250
Boosted weight per planetary vehicle	21,000 lb	21,600 lb	22,500 lb	22,500 lb

*Includes 50-lb capsule bus support equipment plus 120 lb planet scan platform and science booms in 1973 and 1975.
**Includes instruments and DAE only.
***Includes sterilization canister of which 650 lb may remain with the spacecraft.
****650-lb fixed mount camera system replaces scan platform.

4) Limitations on the launch asymptote declination (DLA) are compatible with (a) 2-hour minimum daily launch windows, and (b) launch azimuth ranges of 90 to 115 degrees for Type I transfers in 1973 and 1979, and Type II transfers in 1975 and 1977.

The impact on requirements for the hardware subsystems were primarily a result of the increased science subsystem demands on spacecraft capability. These effects were reflected in additional requirements being placed on the propulsion, power, stabilization, and telecommunications subsystems, as reflected in Tables 2-21 and 2-22, and Figures 2-11 and 2-12, respectively. The science payload weights and power are presented in Table 2-23 and Figures 2-13 through 2-15.

2.4 IMPLICATIONS OF THE 1975-1977-1979 REQUIREMENTS

2.4.1 Purpose

The purpose here was to identify and evaluate the impact of 1975-1977-1979 Voyager Mars performance and design requirements on the 1973 spacecraft system.

2.4.2 Approach

The difference between the 1973 and 1975-1977-1979 requirements indicates that spacecraft changes are required for the 1975-1977-1979 missions. Table 2-24 summarizes the impact of these requirements on the 1973 spacecraft design. Detailed descriptions of the impact of the 1975-1977-1979 requirements on the 1973 spacecraft definition are presented in Volume 3, D2-115002-3, Section 1.1.5.

2.4.3 Conclusions

The following conclusions resulted from assessing the implications of the 1975-1977-1979 requirements:

1) Changes to the 1973 Mars Voyage spacecraft are required for the 1975 mission.

2) For the 1977 and 1979 missions, the fixed-mount high-resolution film camera will have a significant effect on the mission requirements.

Table 2-21: POWER ALLOCATION CHANGES FROM 1973 REQUIREMENTS

	MISSION YEAR			
	1973	1975	1977	1979
Flight Capsule Power (watts d.c.)	200	0	0	0
Science Subsystem Power				
Interplanetary cruise/minimum orbital (watts a.c.	38	58	94	62
Peak orbital power (watts a.c.)	163	223	352	367
Total orbital load (watt-hr)*	591	873	1410	1060
Radio Subsystem Power	223	223	423	423

Maneuver requirements for accommodation of fixed mount imaging (1977-1979), Table 2-22 also apply to power subsystem.

* Orbit period - 12.4 hours

Table 2-22: SPACECRAFT STABILIZATION REQUIREMENTS

- Maneuver Requirements for Accommodation of Fixed Mount Imaging*
    - ± 60-degree roll maneuver on each orbit
    - ± 45-degree pitch maneuver on each orbit
    - ± 45-degree yaw maneuver on each orbit
    - 30-minute inertial hold on each orbit

*Preliminary estimate based on accommodating:

- Viewing cone angles of 90 to 120 degrees (illumination angles of 0 to 30 degrees from terminator)
- Rotational alignment of imaging device with flight path trace on planet surface

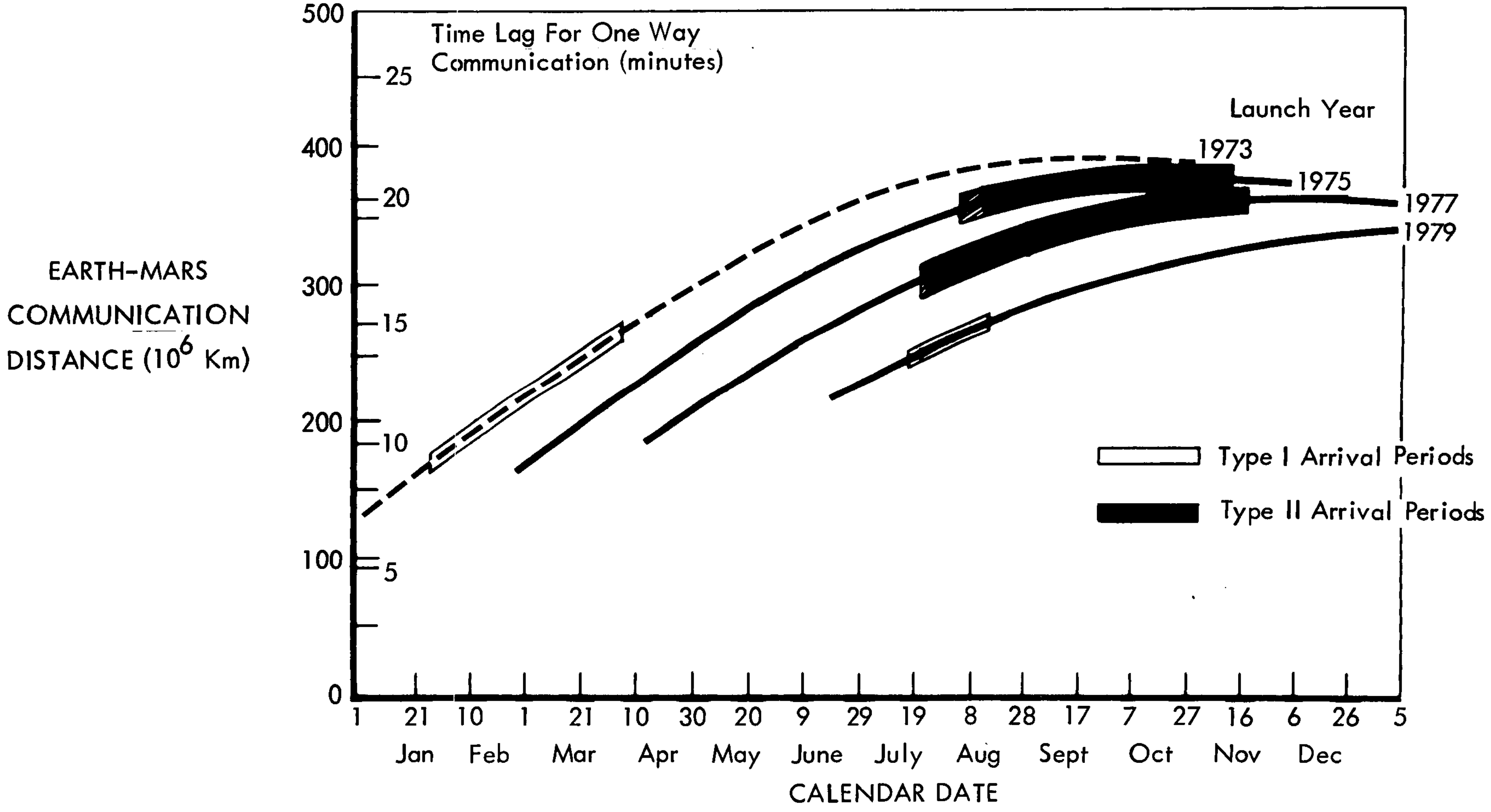


Figure 2-11: EARTH-MARS SEPARATION RANGE

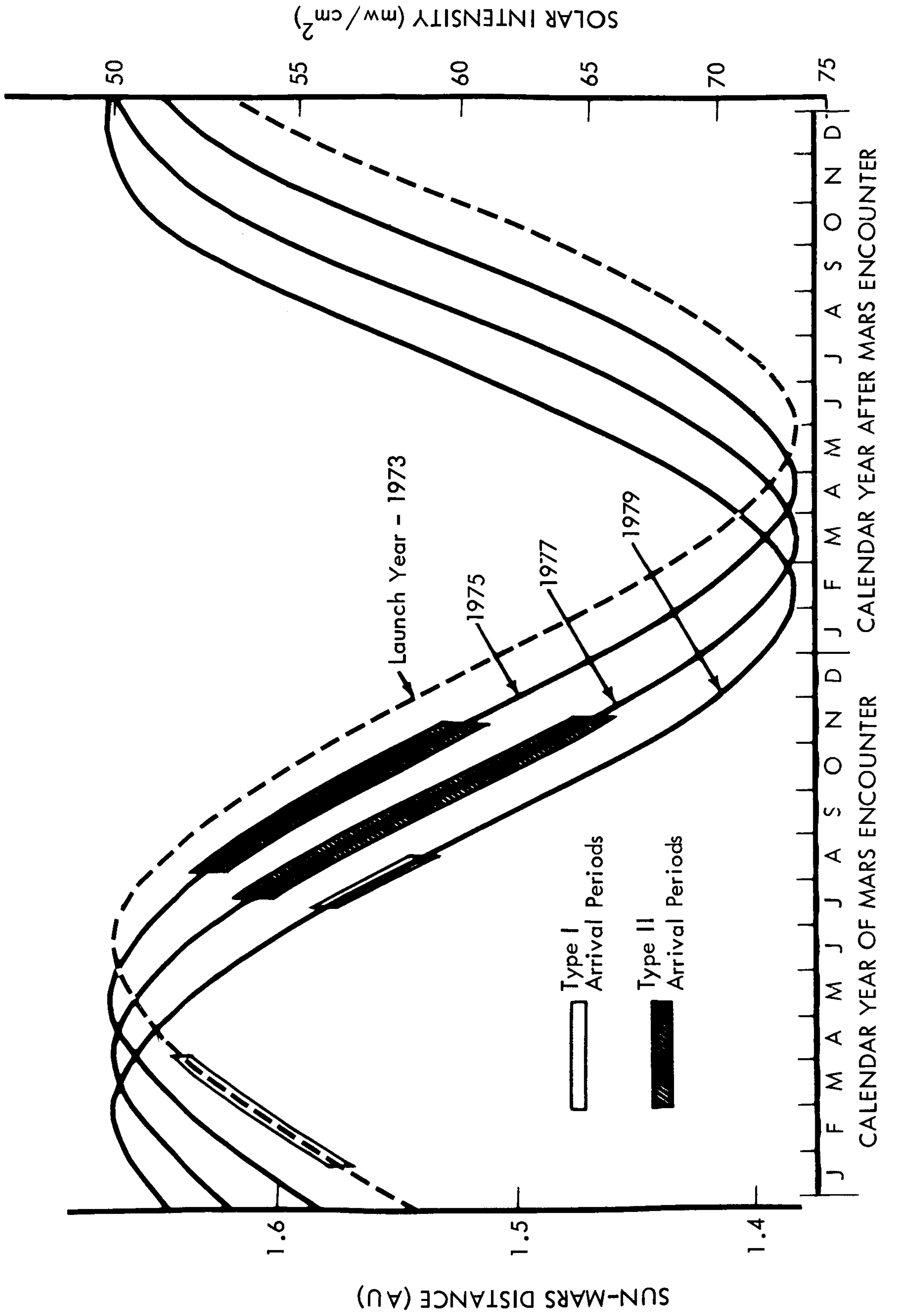


Figure 2-12: SUN-MARS SEPARATION RANGE

Table 2-23: SCIENCE INSTRUMENT WEIGHTS

EXPERIMENTS	INSTRUMENT WEIGHTS (Pounds)		
	1975	1977	1979
Imaging	165	660	660
Broad Band IR Spectrometer High Resolution IR Spectrometer	41	41	41
IR Radiometer	25	25	25
UV Spectrometer	40		
Atmosphere Polarimeter		9	
Cosmic Dust Detector		5	
Subsatellite*	160		160
Bistatic Experiment	40	45	40
Bistatic Radar		20	
Solar Occultation		30	
Gamma Ray Spectrometer	15	20	
Neutron Albedo			30
Meteor Flux		6	
Celestial X-Ray		15	
Ionosphere Sounder	25	25	
Cosmic Ray Telescope	8		
Ion Chamber	3		
Subtotal Science Instruments	522	901	956
DAE	50	60	60
Miscellaneous Support; Power Switch, Elec., Cabling, etc.	20	40	40
Total	592	1001	1056

* Includes:
Electromagnetic probing
Magnetometry
Electric fields
Plasma and charged particles

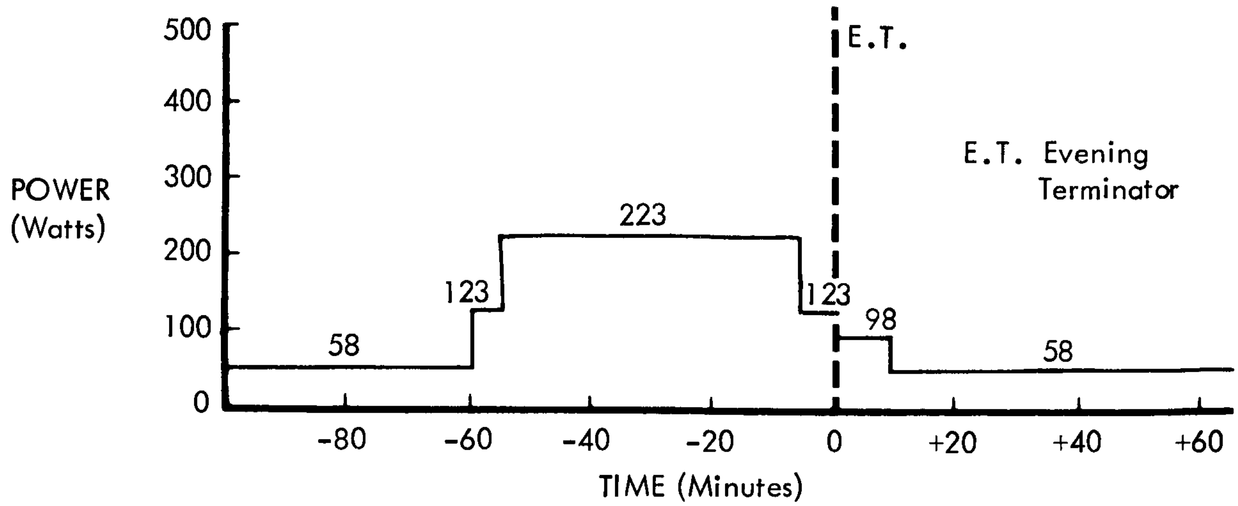


Figure 2-13: SCIENCE PAYLOAD POWER PROFILE — 1975

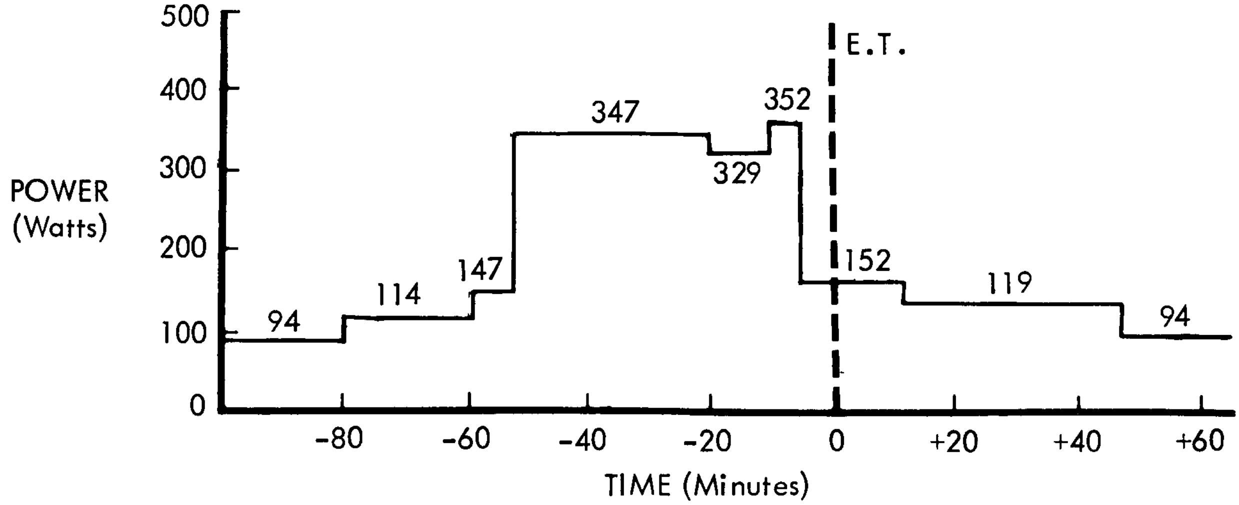


Figure 2-14: SCIENCE PAYLOAD POWER PROFILE — 1977

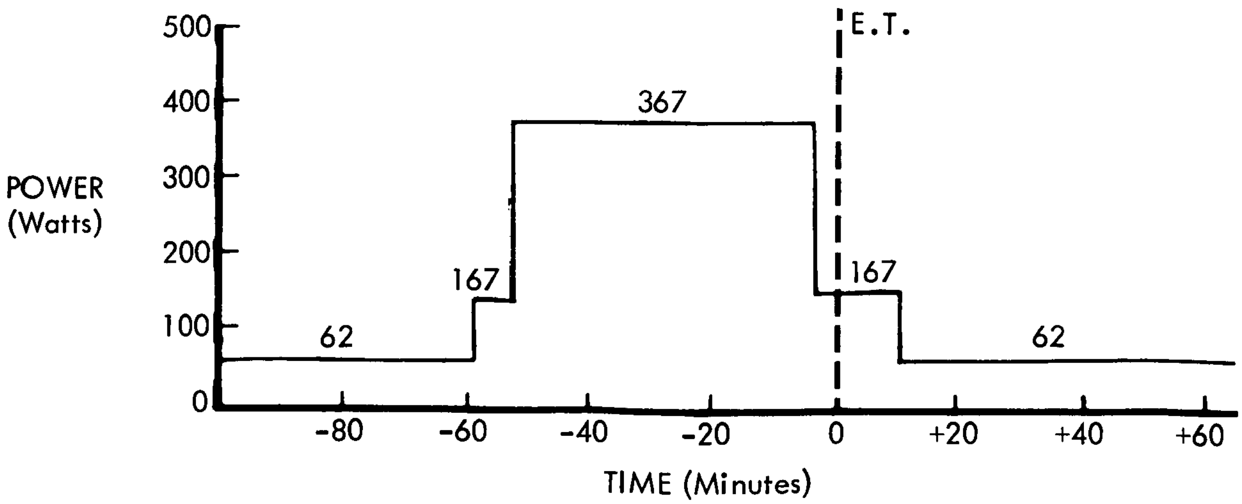


Figure 2-15: SCIENCE PAYLOAD POWER PROFILE — 1979

Table 2-24: IMPACT OF 1975-1977-1979 REQUIREMENTS ON 1973 SPACECRAFT

CHARACTERISTICS/REQUIREMENTS	IMPACT	MISSION APPLICABILITY		
		1975	1977	1979
Mission Definition and Profile	Higher Capsule Weights	X	X	X
Mission Requirements - Operational Life	Increased Operational Life		X	X
Trajectory Parameters - Interplanetary Transfer	Longer Trip Times For Type II Trajectories	X	X	
Trajectory Parameters - Operations Planning				
Trajectory Parameters - Mars Orbit	Shorter Lifetime Allows Lower Periapsis	X	X	X
Target Weight Allocations	Increased Capsule and/or Science Weight	X	X	X
General Design Requirements	Capsule RTG Accommodation	X	X	X
Propulsion Requirements	Increased Propellant Loading	X	X	X
Capsule Support Requirements	Decreased Power Requirement	X	X	X
Data Return Requirements	Increased Earth Return Data Rate		X	X
General Capsule Requirements	Increased Operational Life		X	X
Electrical Power Requirements	Increased for Science Support	X	X	X
Reliability Allocations	Longer Missions Lower Success Prob.		X	X
Spacecraft Stabilization Requirements	Increased RCS Propellant & Off Ref. Times		X	X
Science Payload Requirements	Increased Weight	X	X	X

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3.0 SYSTEM ANALYSES AND TRADE STUDIES

3.0 SYSTEM ANALYSES AND TRADE STUDIES

System analyses and trade studies were conducted to assess and supplement the performance and design requirements for the Mars Voyager spacecraft and missions. They were then used as an input to the spacecraft design and to mission operations planning as applicable. Hardware trades that develop the selection of specific units of equipment are included in Document D2-115002-3, Spacecraft Functional Description.

Aspects of the mission and system were selected for analysis on the basis of critical phases and events during the mission. Nine key system-level studies were chosen. These are:

1) Trajectory Selection Analysis

2) Launch Delay Study

3) Mission Energy Balance Study

4) Encounter Communication Distance Study

5) Capsule-Canister Separation Trade

6) Propulsion Engine Selection Study

7) Orbital Data Rate Analysis

8) Laser Telecommunication Feasibility

9) RF High Data Rate Study

The relationship of these studies to the mission profile is shown in Figure 3.0-1.

A detailed discussion of the purpose, approach, results, and conclusions of each of the above studies is presented in the following sections.

3.1 TRAJECTORY SELECTION ANALYSIS

3.1.1 Purpose

The primary purpose of this study was to assess the requirements imposed on the trajectory parameters to ensure that the baseline orbit can be attained and can satisfy the objectives of the orbital photoimaging experiment.

3.1.2 Approach

The baseline orbit (500 by 18,600 km) was analyzed to determine its: (1) lifetime, (2) ability to satisfy photoimaging experiment objectives, (3) celestial reference occultation characteristics, (4) orbit trim requirements, and (5) orbit insertion requirements. Parametric data for the above were also developed for evaluating orbits other than the baseline orbit. Interplanetary transfer trajectories for the 1973 mission that satisfy the mission and system performance and design requirements

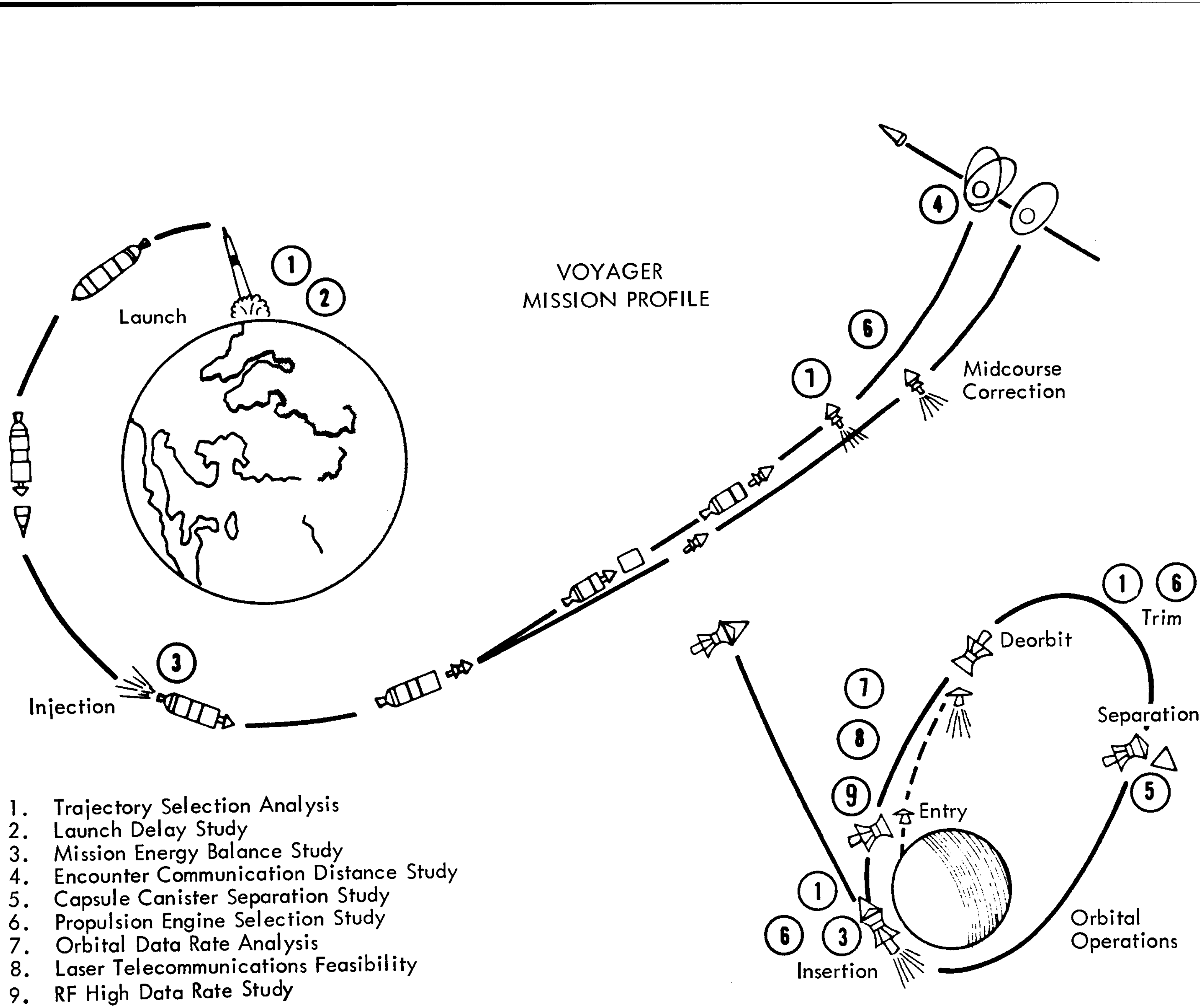


Figure 3.0-1: SYSTEM ANALYSES AND TRADE STUDIES IDENTIFICATION

and accommodate the baseline orbit were then defined. Aiming points and midcourse trajectory correction maneuver requirements for the defined 1973 trajectories were established. The required velocity and attitude control accuracies were determined.

After the 1973 mission trajectory selection was made, interplanetary trajectories for the 1975, 1977, and 1979 missions were defined. The specific 1975-1979 trajectory parameters influencing the 1973 spacecraft design were identified.

3.1.3 Mars Orbit Selection

Orbit selection can be divided into two basic, interrelated considerations: orbit size and orbit orientation. These and other basic orbit parameters are shown in Figure 3.1-1. Orbit size and orbit orientation are governed by:

- Orbit lifetime within the planetary quarantine constraints
- Spacecraft photoimaging objectives
- Propulsion capability for orbit insertion.

The implication of these criteria on orbit size and orientation is discussed below.

3.1.3.1 Orbit Lifetime - 1973

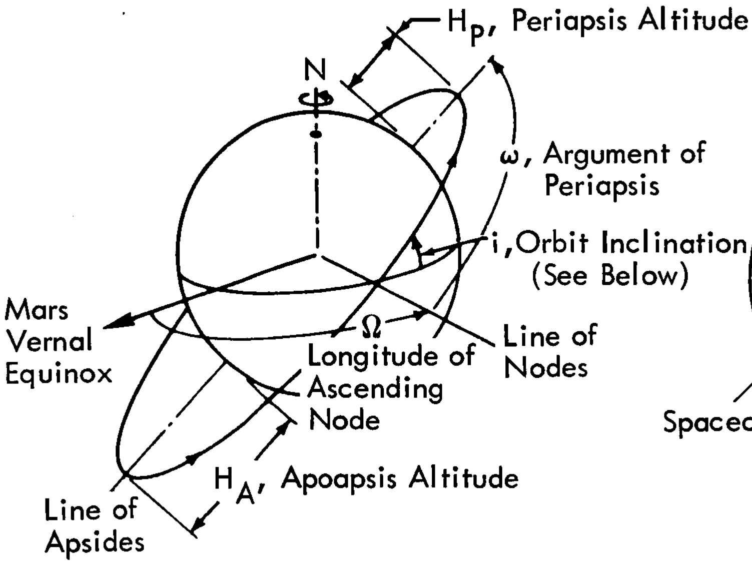
Planetary quarantine requires that the probability of impacting Mars due to orbit decay before 1984 shall be less than 8×10^{-6} (see Figure 3.1-2). To determine the effect of this requirement on orbit size, an orbit lifetime study was conducted considering:

- Atmospheric drag
- Orbit insertion errors
- Astronomical perturbations.

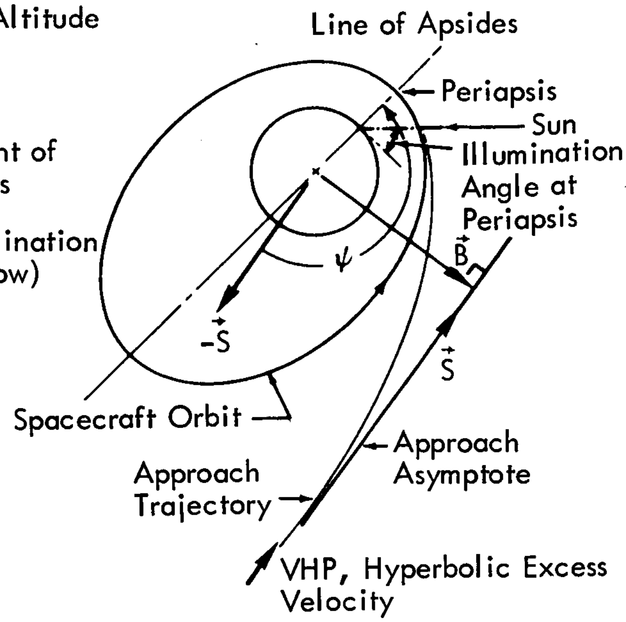
Atmospheric Drag and Orbit Insertion Errors -- The most important factor affecting the lifetime of Voyager orbits around Mars is atmospheric drag. This drag is a function of the atmosphere model. The results presented here are based on an atmosphere model provided by MSFC. For this model, the maximum of the 99% confidence envelope was used. For a given orbit, the lifetime is directly proportional to the ballistic coefficient. Figure 3.1-3 shows the ratio of lifetime to ballistic coefficient as a function of periapsis and apoapsis altitude. These results are based on a nonrotating atmosphere. The effect of atmospheric rotation on orbit lifetime is not significant. Moreover, for posigrade orbits, a nonrotating atmosphere will yield a lower orbit lifetime than a rotating one and, therefore, is conservative.

To ensure that the planetary quarantine requirement is not violated, a probability analysis has been conducted. The analysis combined the probability of successfully conducting an orbit trim maneuver with a set of initial dispersions in orbit parameters following insertion. From this analysis, periapsis altitude modification required to satisfy the planetary quarantine constraint can be determined. The dispersions assumed for initial orbit parameters were:

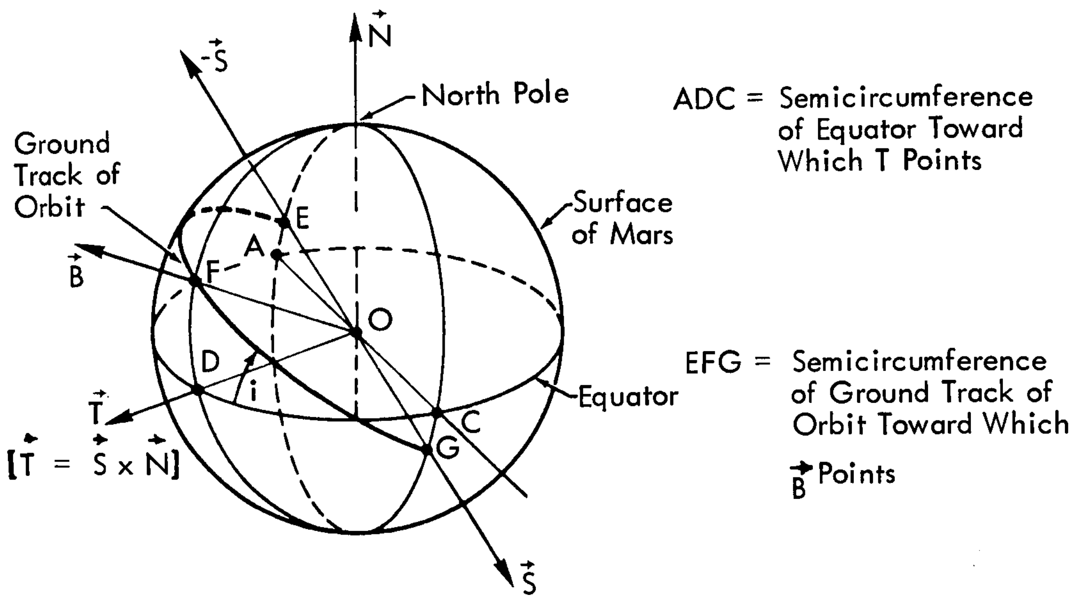
ORBIT SIZE AND ORIENTATION



APPROACH AND INITIAL ORIENTATION



ORBIT INCLINATION



Inclination, i, is the dihedral angle between the orbit plane and the equator. It is measured positive counterclockwise from half-plane O-ADC to half-plane O-EFG.

Figure 3.1-1: DEFINITION OF BASIC ORBIT PARAMETERS

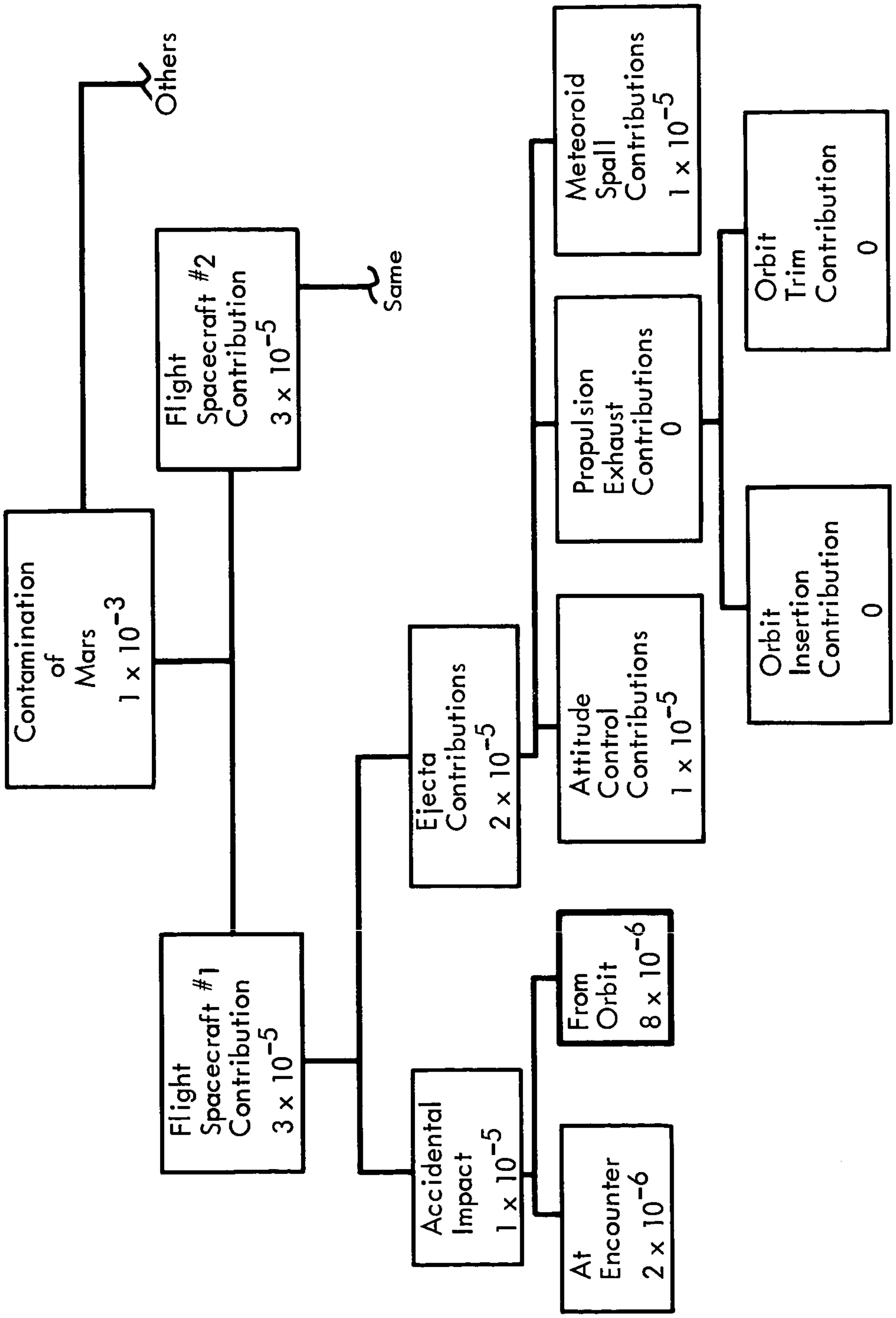


Figure 3.1-2: ALLOCATION OF PROBABILITY OF CONTAMINATION—PLANETARY QUARANTINE

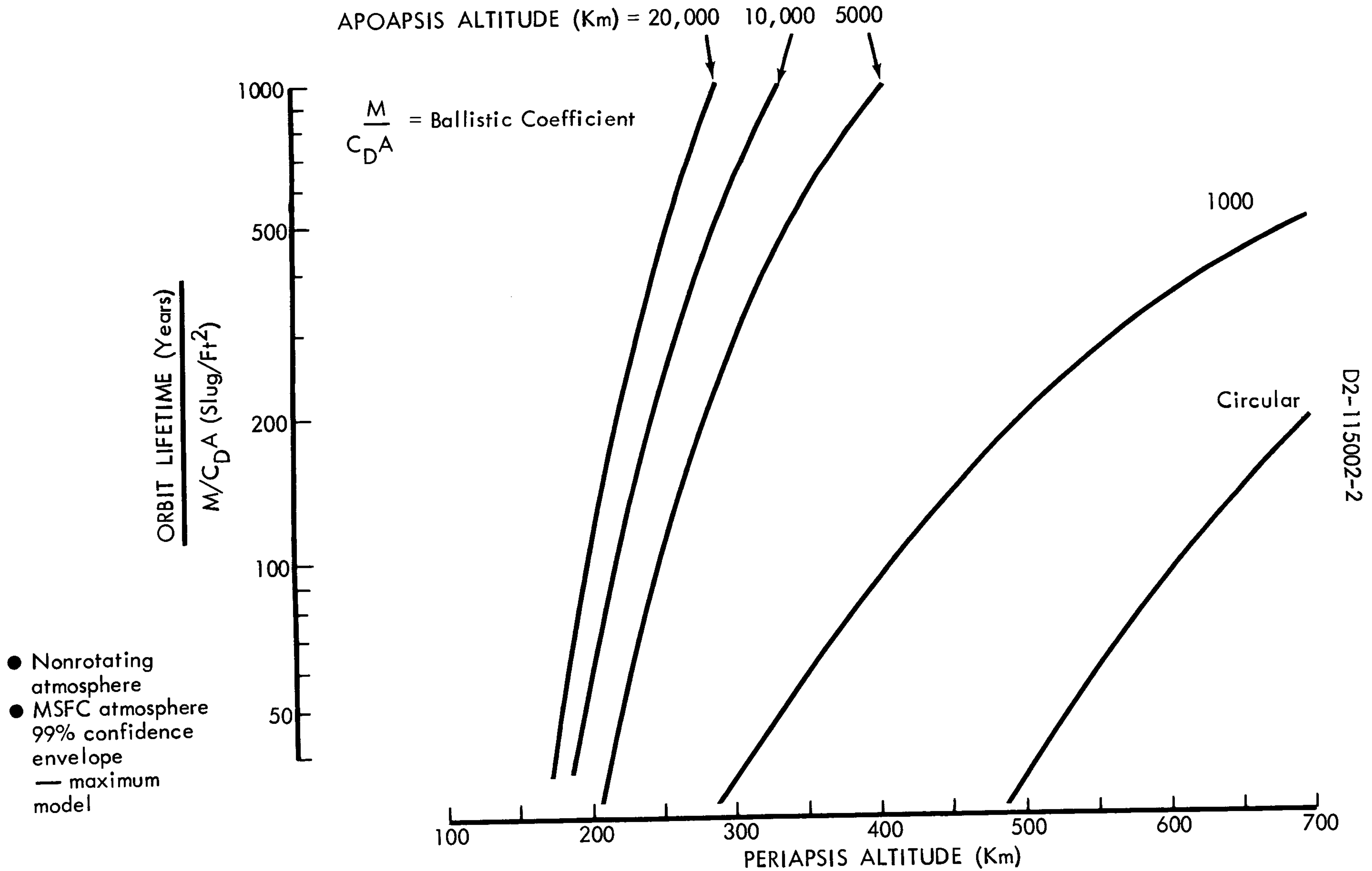


Figure 3.1-3: ORBIT LIFETIME AERODYNAMIC EFFECTS

Periapsis altitude	± 45 km 1 σ
Orbit period	± 0.409 hours 1 σ

The probability of successfully accomplishing the orbit trim maneuver was estimated to be 0.99.

A table for the bivariate normal distribution was used to determine the number, n, of standard deviations associated with the probability that the required lifetime is met should the orbit trim maneuver fail. The value of n, to accommodate the 0.99 probability of orbit trim success and the 8×10^{-6} planetary quarantine allocation, is 3.91.

The implications of this n value for the lifetime of the baseline orbit (i.e., periapsis altitude = 500 km; period = 12.4 hours) can be determined from Figure 3.1-4. This figure relates the orbit lifetime parameter to the periapsis altitude for a given orbit period. The orbit lifetime parameter, as discussed above, is the ratio of the orbit lifetime to the ballistic coefficient. The capsule forward sterilization canister is separated after orbit insertion. Consequently, the governing ballistic coefficient from orbit lifetime considerations is that of the canister. For a 6000-pound flight capsule, that parameter is estimated as 0.01 slugs/ft^2. Because the canister must stay in orbit for 11 years (1974-1984), the orbit lifetime parameter is approximately 1000. From Figure 3.1-4, the minimum periapsis altitude for a 12.4-hour orbit having a life parameter of 1000 is less than 300 km. However, as indicated previously, dispersions of 3.91σ in periapsis altitude and orbit period must be accommodated to ensure a probability of planetary contamination of 8×10^{-6} for an 0.99 reliable orbit trim maneuver. Consequently, as indicated in the figure, a nominal periapsis altitude of 478 km is required. This is slightly lower than the 500 km periapsis of the baseline orbit. The baseline orbit therefore is conservative from aerodynamic drag considerations.

Astronomical Effects -- The baseline orbit was shown to be safe, considering aerodynamic drag only. This must now be checked considering both drag and astronomical disturbances to the orbit.

For orbits around Mars, the primary astronomical perturbation is caused by the Sun. The gravitational anomalies of Mars must also be considered, however, because they influence the rotation of the orbit with respect to the Sun. Astronomical perturbations are not simple, and general relationships (e.g., Figure 3.1-3) cannot be obtained. However, for any specific orbit, it is possible to obtain a time history of periapsis altitude (and all of the orbital elements as well) from which orbit lifetime is determined. Such studies were conducted for the baseline and other candidate orbits. The analysis accounted for the effect of: multiple perturbing bodies, i.e., Mars and the Sun, gravitational anomalies of Mars, atmospheric drag, and solar radiation pressure.

Lifetime predictions for any specific orbit around Mars are limited by knowledge of the Martian gravity field. The gravity field of Mars accelerates the orbiting vehicle. This acceleration is a function of: (1) the orbit around Mars, (2) the mass of Mars, (3) the oblateness, J_2, of Mars, (4) the "pear shape," J_3, of Mars, and (5) higher order terms. Items (1), (2), and (3) are relatively well known; items (4) and (5) are not known. Fortunately, for orbits with small periapsis altitudes, item (5) can be neglected.

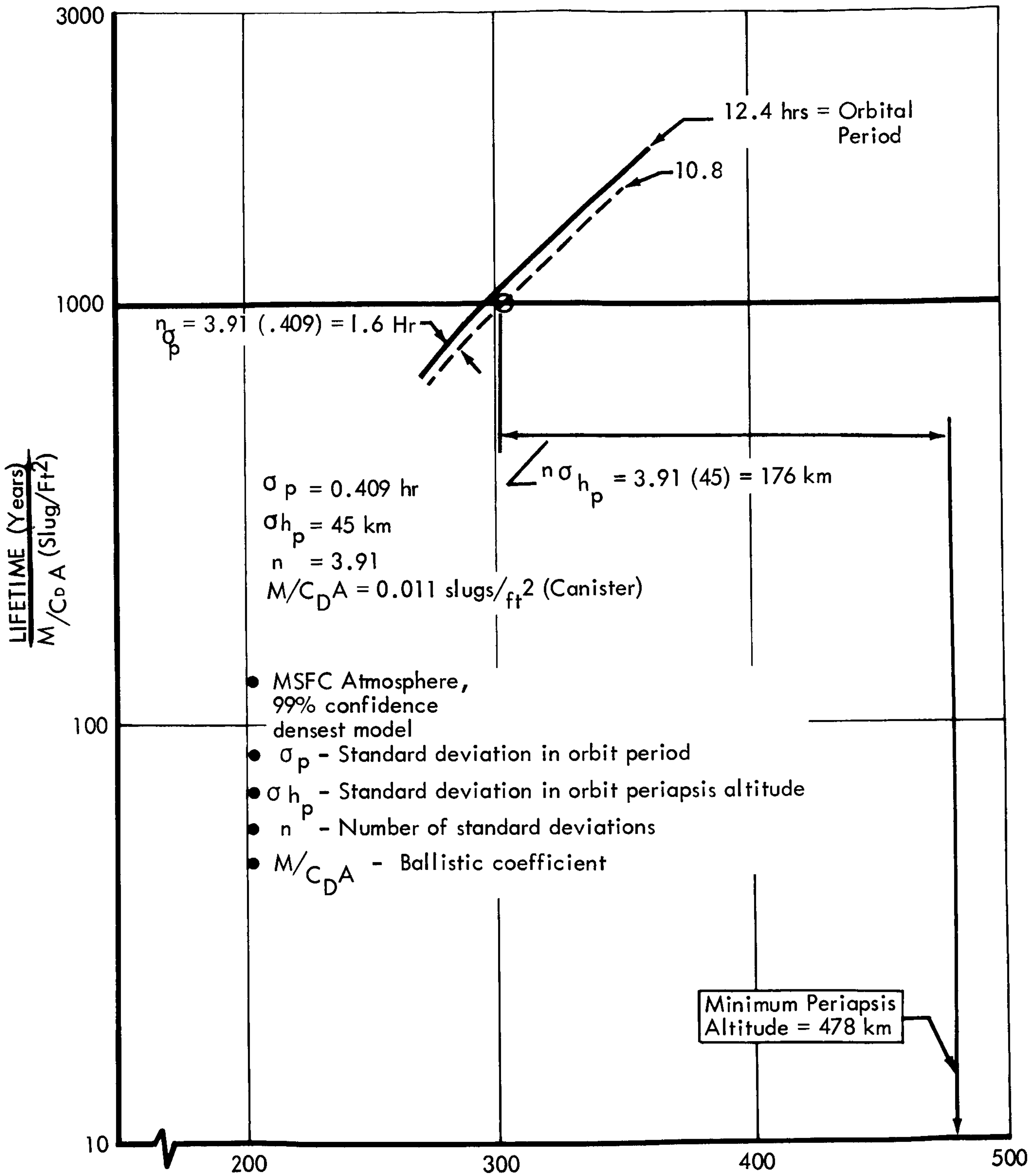


Figure 3.1-4: DETERMINATION OF MINIMUM PERIAPSIS ALTITUDE

The coefficient, J_3 (the "pear-shape" coefficient), results in a secular change in eccentricity and hence in periapsis altitude. The magnitude of J_3 has been estimated to be 1.5×10^{-5} by analogy with Earth, but its sign cannot be determined.

Lifetime results for the baseline orbit, including the effects of astronomical perturbations, are shown in Figure 3.1-5.

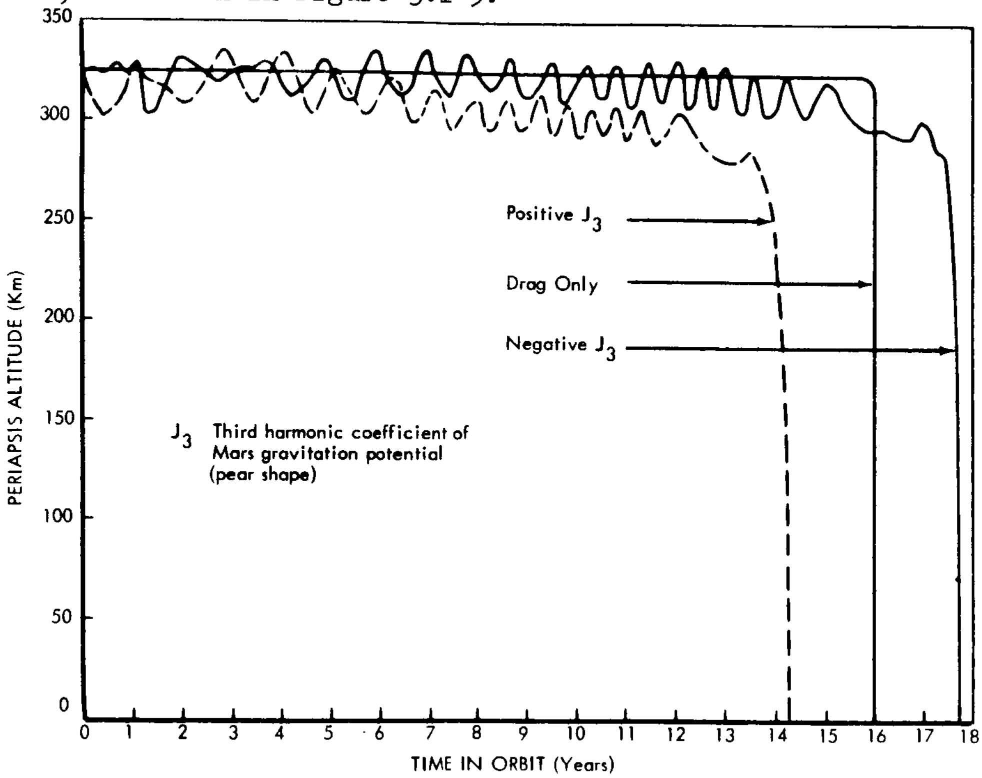


Figure 3.1-5 : HISTORY OF PERIAPSIS ALTITUDE FOR THE DISPERSED DESIGN ORBIT

The initial periapsis selected here was 500 km minus 3.91σ, or 324 km; the initial period was also reduced by $3.91\sigma_p$. From the figure, the conclusion is that the baseline orbit is safe in the presence of both drag and astronomical perturbations; the periapsis altitude of 500 km, therefore, is conservative.

Because of the aforementioned orbit insertion errors, the initial planetary vehicle around Mars will be dispersed with respect to the desired baseline orbit. An orbit trim maneuver, therefore, will be required to adjust the initial orbit into the baseline orbit. Two orbit trim maneuvers with a combined ΔV of 150 m/sec were allocated to satisfy the orbit-adjusting requirement. Consequently, an analysis was conducted to assess the adequacy of this preliminary allocation.

It was decided to use the orbit trimming capability to control the radius of periapsis, the radius of apoapsis, the longitude of the ascending node, and the argument of periapsis. Control of the apsidal radii (i.e., periapsis and apoapsis) is important, because these radii establish the orbit lifetime and determine the period. The orbit period, as noted above, controls imaging coverage and the illumination angle at, and near periapsis. Desired imaging location and illumination angle also require control of the longitude of the ascending node and the argument of periapsis.

An orbit is defined by six parameters. A maneuver at a specified time can only control three parameters, because only the spacecraft velocity, which has three components, is modified by the maneuver. The desired four parameters are controlled by performing two trim maneuvers. The first trim maneuver corrects radius of periapsis (R_A), radius of apoapsis (R_P), and argument of periapsis. The second maneuver corrects R_A, R_P, and longitude of the ascending node.

The choice of the true anomalies at which the two trim maneuvers are made is used to minimize the ΔV required for orbit trim. For the baseline orbit, the true anomalies that result in minimum orbit trimming ΔV are given below:

	Parameter Corrected	Trim True Anomaly (degrees)
First Trim	R_P, R_A, ω	110 to 130
Second Trim	R_P, R_A, Ω	210 to 240

The three σ errors of the trimmed orbit, following two orbit trims performed to minimize trim ΔV, are summarized in Table 3.1-1 for three Mars arrival dates:

Table 3.1-1: ERRORS AFTER ORBIT TRIM MANEUVER
1973 ARRIVAL DATE

3σ Errors In	Early	Middle	Late
R_P (km)	11	7.1	6.5
T (sec)	65	39	37
ω (deg)	4.6	4.6	3.4
Ω (deg)	0.04	0.04	0.04
i (deg)	3.3	2.8	3.5
$3\sigma\,\Delta v$(m/sec)	211	153	131

The table indicates that the trim ΔV allocation of 150 m/sec can accommodate March and April arrival dates. Early arrival dates (i.e., February) will require increasing the trim ΔV allocation by approximately 60 m/sec. The planetary vehicle launch weight penalty associated with such an increase will be significant (approximately 400 pounds).

3.1.3.2 Spacecraft Photoimaging Objectives

Photoimaging objectives of the Voyager spacecraft are:

1) Good resolution (photography from low altitudes in the vicinity of periapsis).

2) Coverage of a wide latitude band.

3) Illumination of target sites to provide good contrast (preferred illumination angle of 15 degrees).

It is desirable to control the periapsis location so that these objectives are best satisfied throughout the mission. As a basis for this study, and for subsequent trades, the baseline orbit specified in Table 3.1-2 was considered. Figure 3.1-6 illustrates the ground track of the first two revolutions of the baseline orbit around Mars. The band of favorable illumination for photography covers a latitude range from -5 to +30 degrees on the first day. While in this latitude-illumination band, the spacecraft altitude is always less than 1500 km, i.e., below three periapsis altitudes. This illumination band is centered at 0 degree longitude. A shift in initial longitude coverage is obtained by changing the arrival time in terms of hours. If the arrival were at 19:00 GMT, the favorable illumination conditions would be centered in the area of 75 degrees longitude (above Syrtis Major).

TABLE 3.1-2: BASELINE ORBIT DEFINITION

Periapsis Altitude (km)	500
Period (hr)	12.4
Apoapsis Altitude (km)	18,590
Inclination to Martian Equator (deg)	+45
Time of First Periapsis Passage (date) (hr, GMT)	April 4, 1974 0:00
Longitude of Ascending Node (deg)	289.4
Argument of Periapsis (deg)	167.5

Figure 3.1-7 illustrates the illumination-time history of the baseline orbit. The latitude coverage is good. A band between +29 and -45 degrees is covered within 120 days. Also, the periapsis point progresses across that latitude band at favorable illumination angles over the 180-day mission. During this time period, the optimum illumination angle of 15 degrees is attained at altitudes between 500 (periapsis) and 735 km.

- Launch Date: August 25, 1973
- Arrival Date: April 4, 1974
- Periapsis Altitude = 500 Km
- Orbit Period = 12.4 hours
- Inclination = +45 degrees

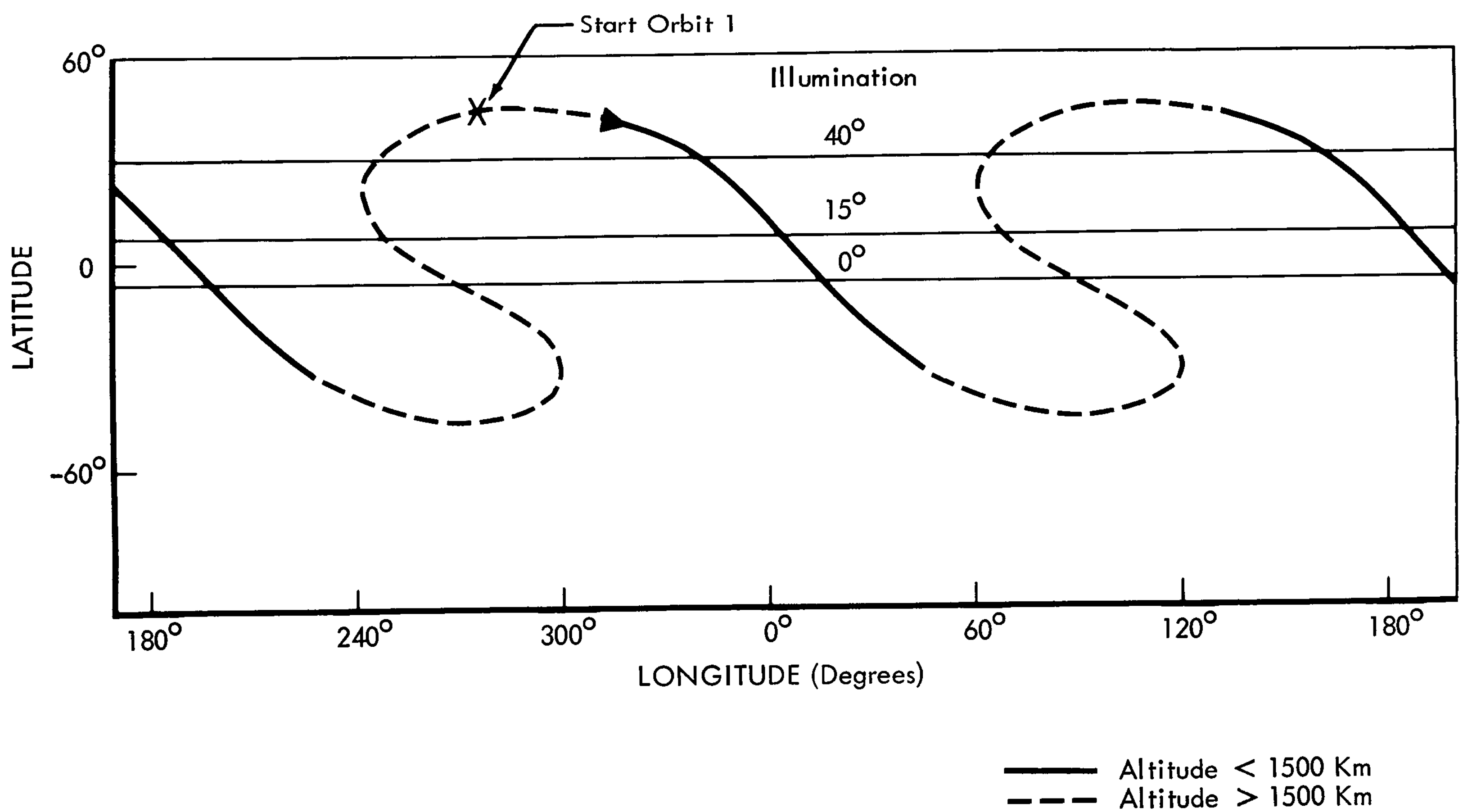


Figure 3.1-6: BASELINE ORBIT ILLUMINATION BAND ON FIRST DAY

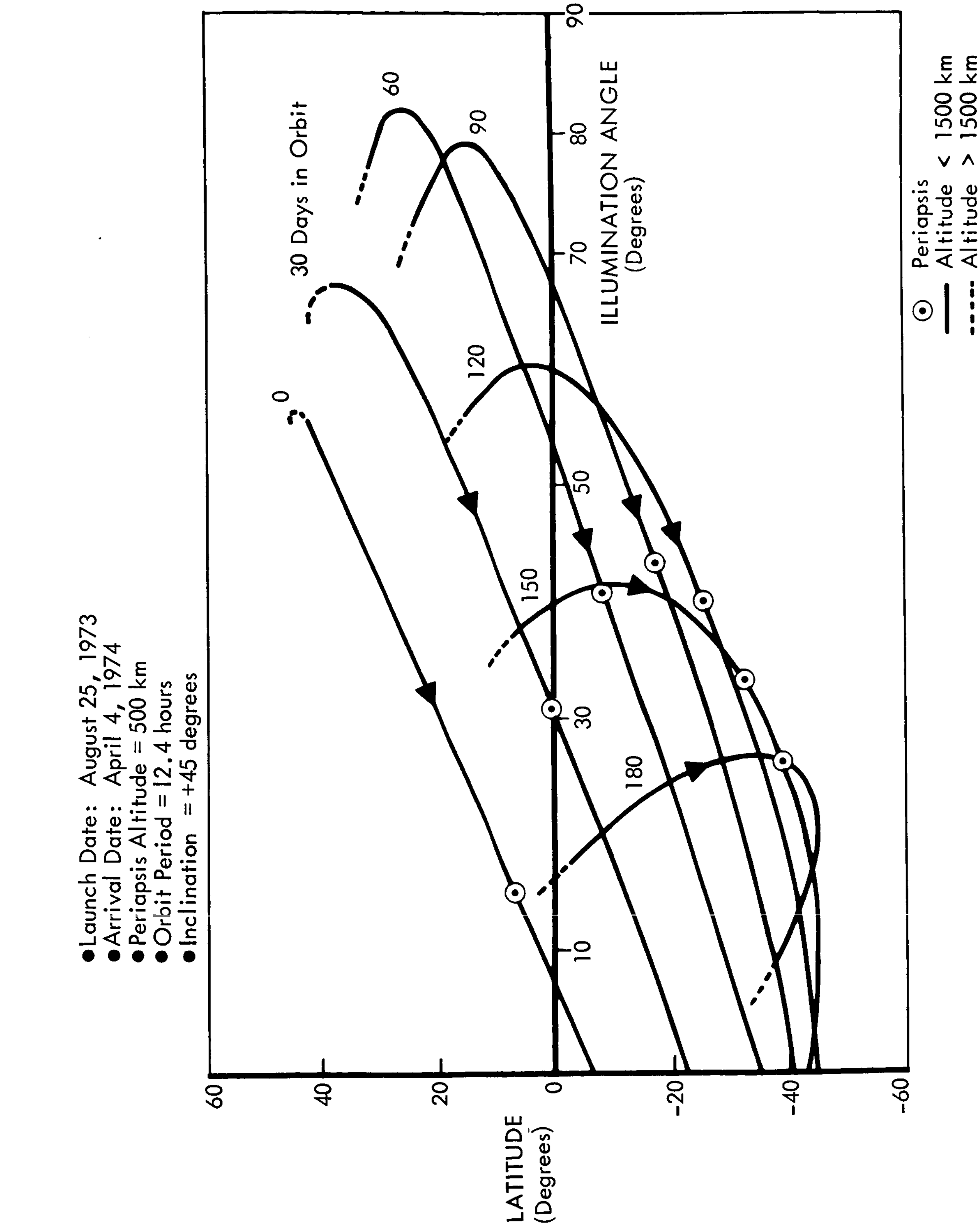


Figure 3.1-7: BASELINE ORBIT PHOTOGRAPHIC COVERAGE

The nodal separation between adjacent orbit ground tracks is 2.5 degrees. If photographs covering 100 x 100 km of the planet surface are considered, contiguous longitudinal photographic coverage at the Equator can be achieved in approximately 74 days.

The ability in the baseline orbit to observe the wave of darkening is shown in Figure 3.1-8. The loci of periapsis and the 15-degree illumination point during an 180-day orbital mission are given.

Orbits that are favorable from photoimaging considerations, i.e., illumination and coverage, may violate the Sun occultation constraint (see Section 2.2), result in undesirable Canopus occultations and be free of Earth occultations which are desirable for the conduct of atmospheric determination experiments. Figure 3.1-9 presents the solar occultation profile of the baseline orbit. Sun

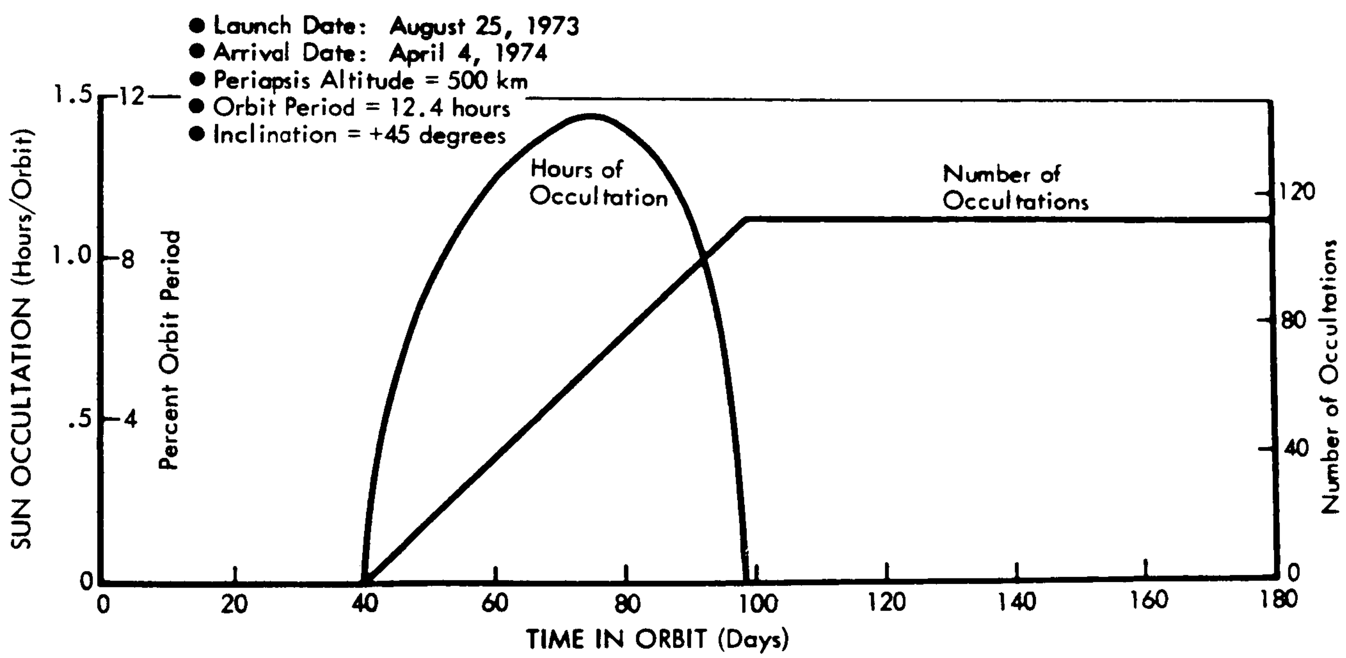


Figure 3.1-9: BASELINE ORBIT SUN OCCULTATION

occultation occurs when the spacecraft-Sun line-of-sight crosses the limb of Mars. The design orbit complies with the requirements that no Sun occultation occurs within the first 30 days of the mission, and that no Sun occultation, thereafter, exceeds the smaller of 1.5 hours or 12.5% of the orbit's period.

The Earth occultation profile for the baseline orbit is presented in Figure 3.1-10. A series of short Earth occultations occurs early in the mission, benefitting the Earth occultation experiment. Longer occultation periods (1.3 hours maximum) occur in the middle of the mission and permit repetition of the experiment over a wider band of Martian latitude. Figure 3.1-11 shows the latitude of occultation as a function of mission duration.

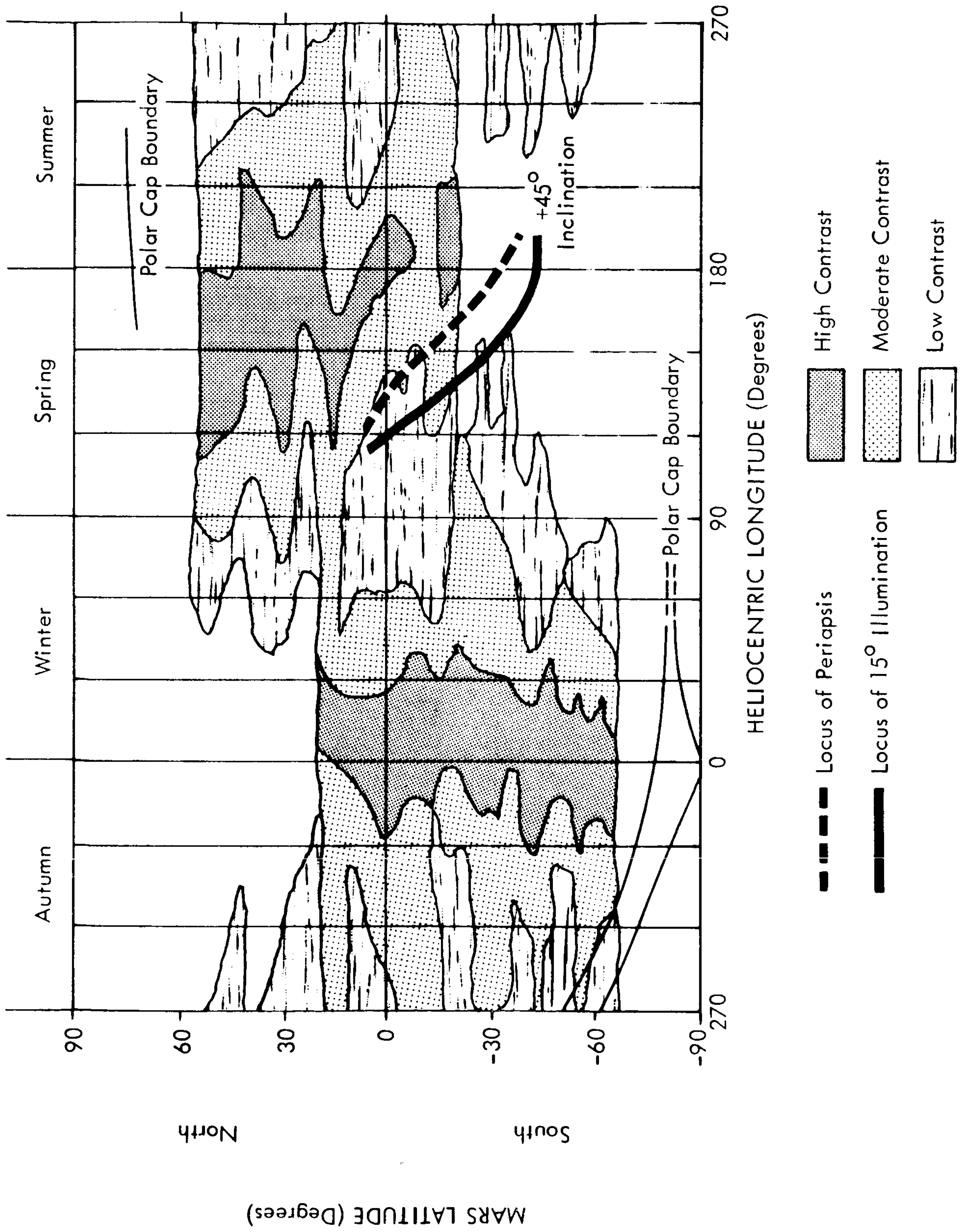


Figure 3.1-8: BASELINE ORBIT OBSERVATION OF WAVE OF DARKENING

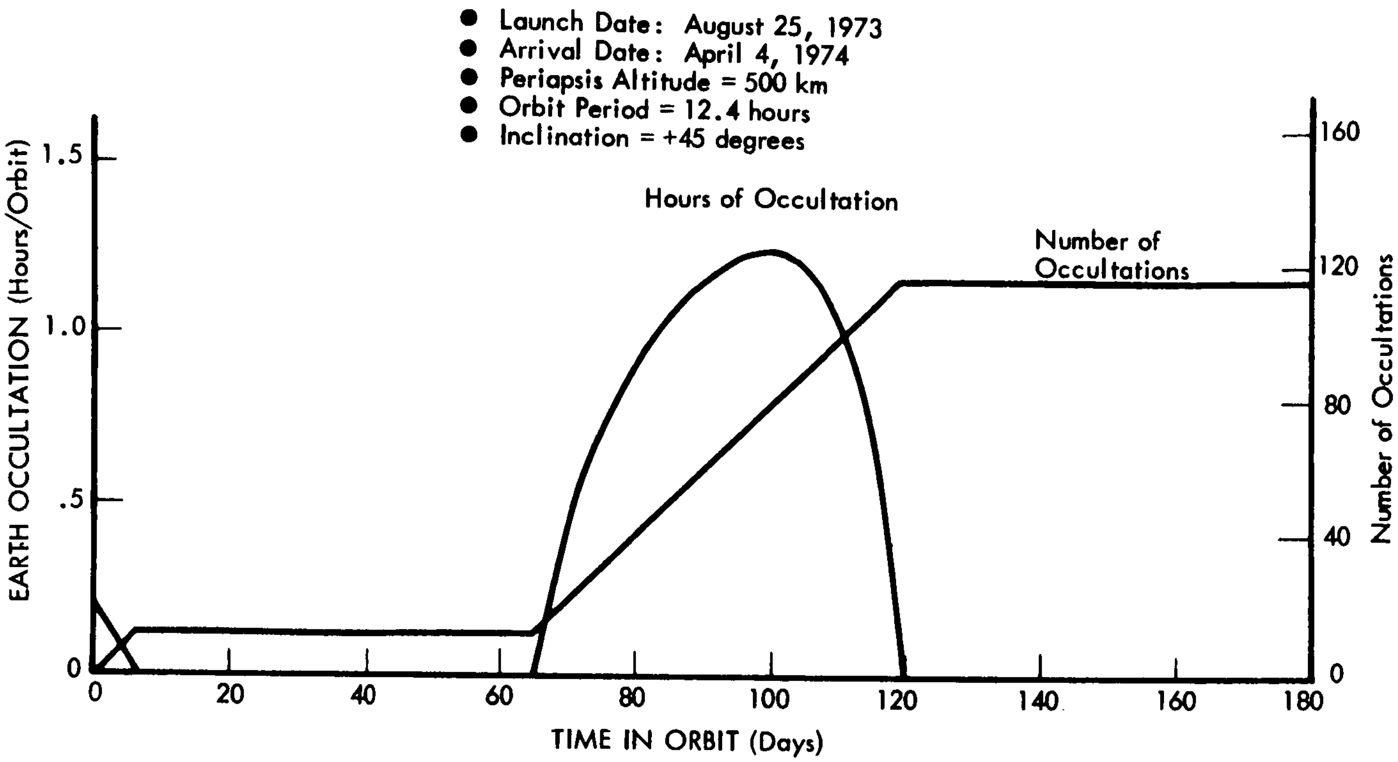


Figure 3.1-10: BASELINE ORBIT EARTH OCCULTATION

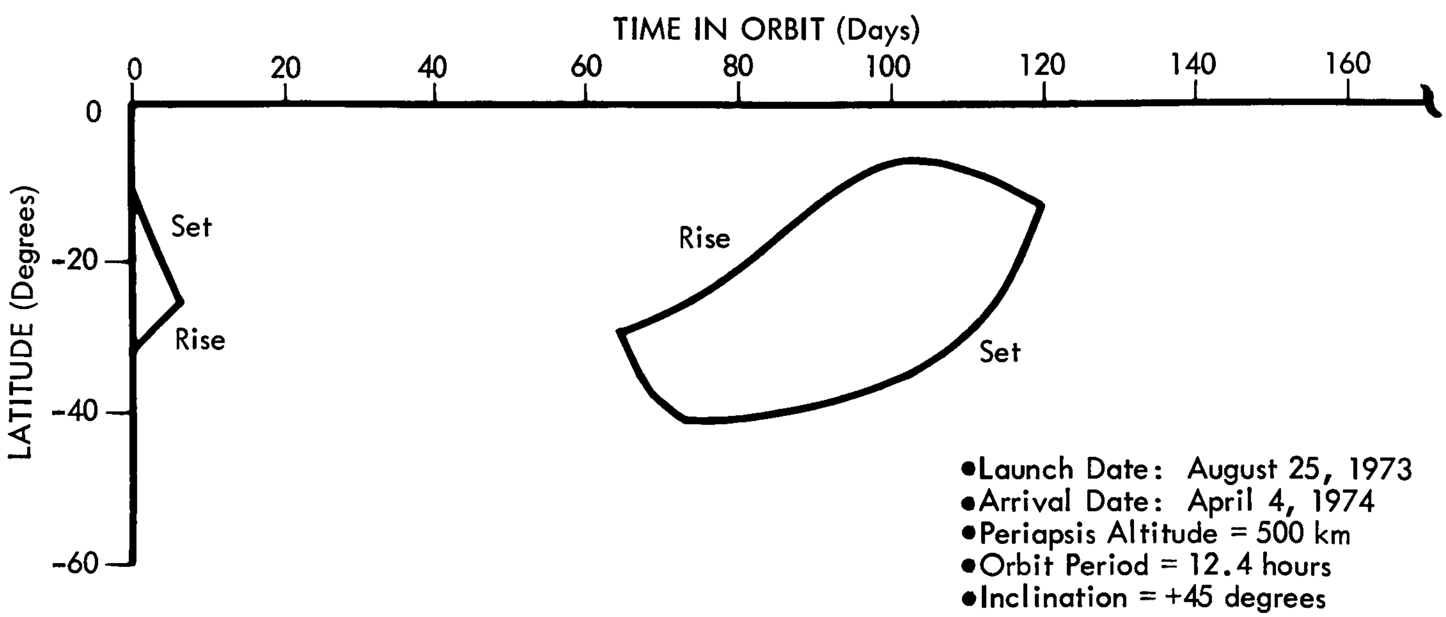


Figure 3.1-11: RISE AND SET LATITUDES BASELINE ORBIT EARTH OCCULTATIONS

To maintain celestial reference throughout the mission, it is desired, though not required, to avoid Canopus occultation or Canopus sensor interference.

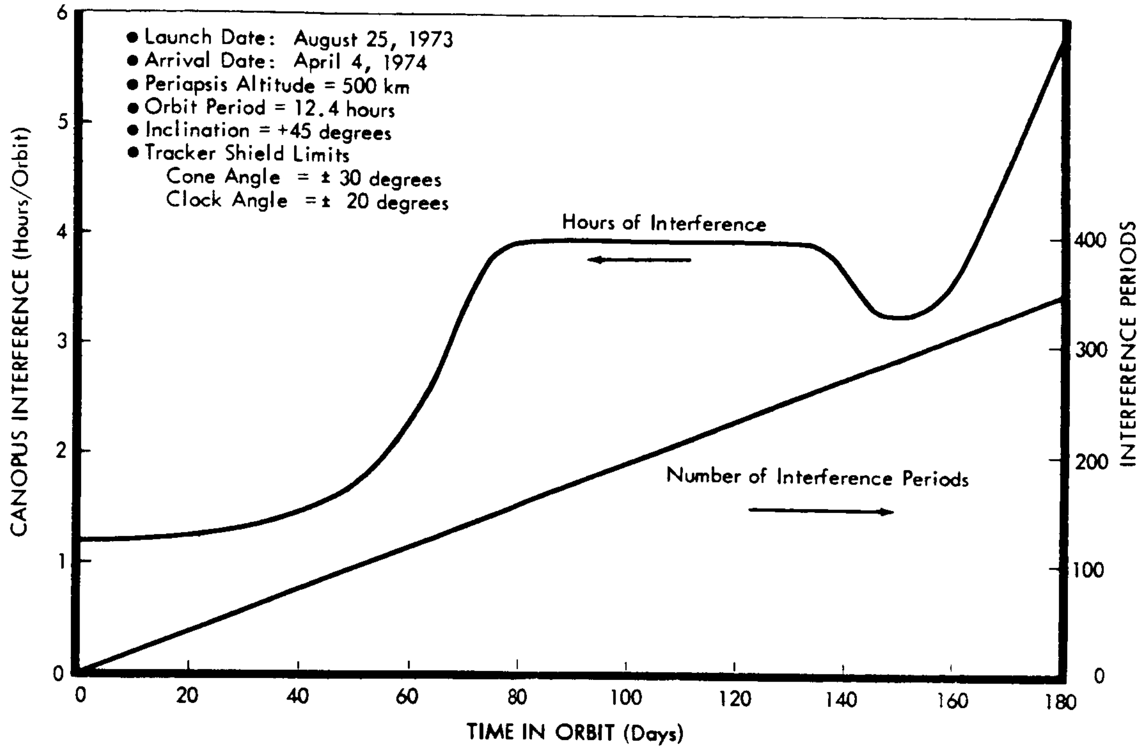


Figure 3.1-12: BASELINE ORBIT CANOPUS INTERFERENCE

Figure 3.1-12 presents the Canopus interference history during the 1973 mission for the baseline orbit. Note that no Canopus occultation occurs during the 180-day mission, but that on each orbit there is a period when the illuminated portion of Mars is within the field of view of the Canopus sensor.

The above results indicate that the baseline 500 by 18,600 km orbit is acceptable from preliminary photoimaging considerations, and Sun, Earth, and Canopus occultation considerations. Additional trades were then conducted to determine whether other orbits exist that are superior to the baseline orbit in the above considerations. These studies were aimed at selecting orbit period/inclination combinations that result in favorable illumination angles at or near periapsis throughout the 180-day orbital mission.

It is desirable to have the periapsis altitude as low as possible to maximize imaging resolution. However, the periapsis altitude must be sufficiently high to ensure adequate orbit lifetime when uncertainty in Mars atmosphere and orbit insertion and trim errors are considered. It has been shown that an altitude of 500 km is near the minimum allowable. This is the value used for the following orbit studies. Consequently, there are only two parameters to be varied: orbit orientation, and apoapsis altitude. Orbit orientation is completely defined by the specification of launch and arrival dates, illumination at initial periapsis, and orbit inclination.

Maintenance of relatively constant illumination at periapsis over long periods of time requires that the relative motion between the line of apsides (i.e., orbit's major axis) and the Mars-Sun line be minimized. This can be accomplished by forcing the vector sum of the motion of the orbit line of nodes and line of apsides to match the motion of the Sun as closely as possible. Typical rates of apsidal rotation, $\dot{\omega}$, and nodal regression, $\dot{\Omega}$, are given in Figure 3.1-13.

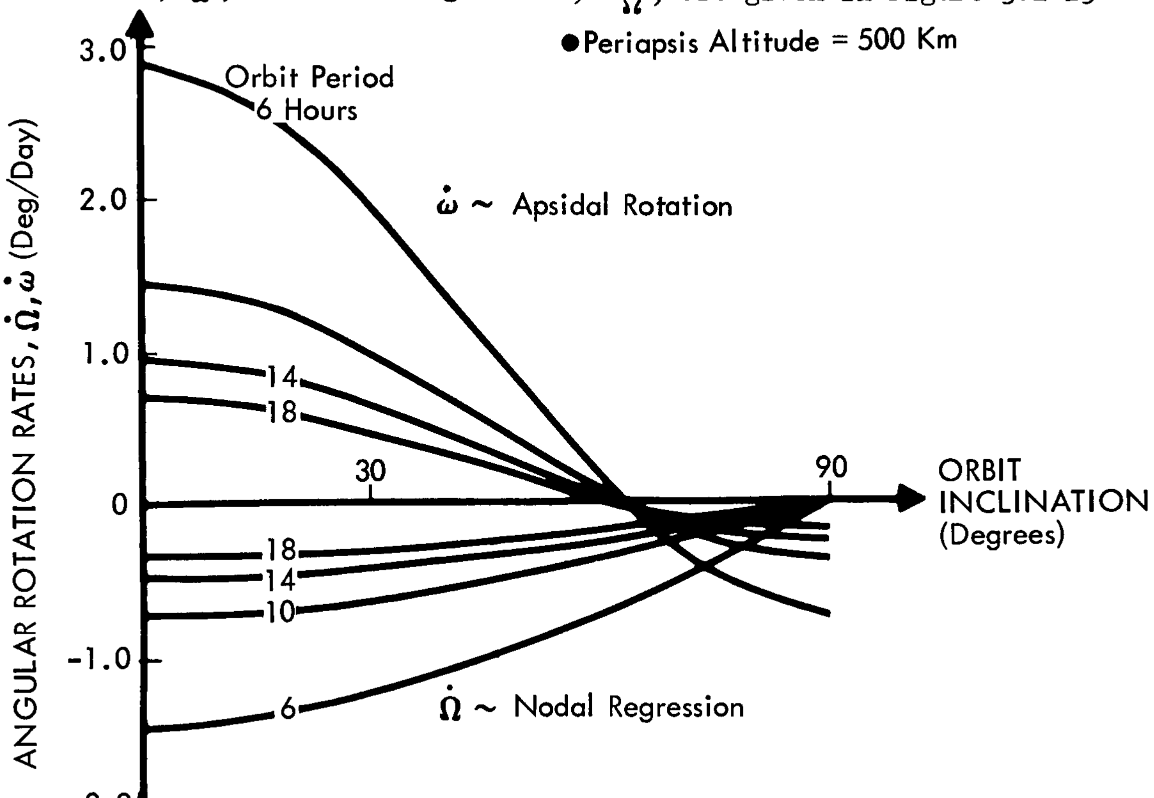


Figure 3.1-13: ORBIT ANGULAR ROTATION RATES

The nodal regression, $\dot{\Omega}$, is negative for posigrade (direct) inclinations. The apparent motion of the subsolar point on Mars is approximately 0.52 deg/day. Consequently, to keep pace with the Sun, the natural apsidal rotation must be large and positive. To obtain this high apsidal rotation, low orbital inclinations are required. To identify the approximate orbit period/inclination combinations that minimize the daily change in periapsis illumination, periapsis motion is resolved into two components, $\dot{P}$ and $\dot{N}$. These two components are parallel and normal, respectively, to the Sun's motion at the beginning of the mission. They are shown as a function of orbit period and inclination in Figure 3.1-14. The data are based on an arrival date of April 4, and an initial periapsis illumination angle of 15 degrees from the evening terminator. If $\dot{P}$ is selected to match the rotational rate of the Sun, i.e., 0.52 deg/day, the corresponding values of $\dot{N}$ are defined. The locus of this value of $\dot{P}$ is superimposed on the $\dot{N}$ curves in Figure 3.1-14(a). It defines the inclination/period combinations that tend to minimize the change in illumination angle early in the mission.

The above discussion pertained to posigrade orbits. Alternatively, the planetary vehicle may be inserted into a retrograde orbit. This results in a sign change in $\dot{\Omega}$. Consequently, high-inclination orbits with relatively constant periapsis illumination are obtained.

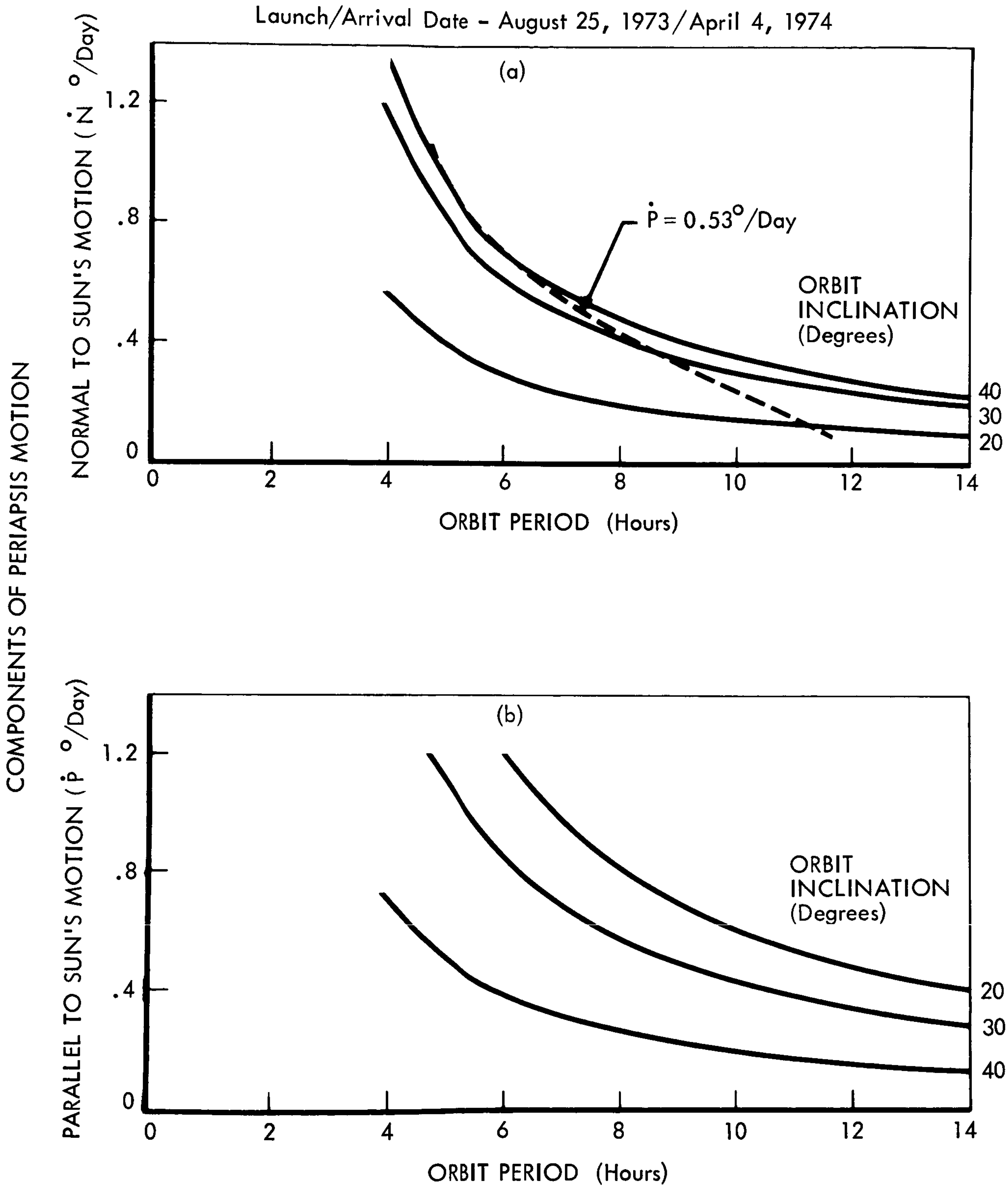


Figure 3.1-14: PERIAPSIS MOTION RELATIVE TO SUN

From the above, it is possible to select orbit period/inclination combinations that minimize the initial rate of change in illumination angle. However, the rate of change of illumination angle at periapsis will change with time because the direction of the Sun's motion, relative to Mars equator, varies with time. Because the illumination at periapsis should be constant over the 180-day mission, best photoimaging results would be obtained at altitudes other than periapsis. Data were computed that enable evaluation of altitude/illumination tradeoffs. Table 3.1-3 lists periapsis illumination angle, and periapsis latitude/time histories for orbit inclinations of 30, 45 and 120 degrees. In addition, the time histories of latitude and altitude for a 15-degree illumination also are given. The data are shown for orbits that result in the least illumination angle variations during a 180-day mission. The data indicate that retrograde orbits of about 120 degrees have near-constant illumination angles at periapsis throughout the mission. They would appear to be the best choice from illumination considerations. However, retrograde orbits are difficult to obtain because of apsidal rotation requirements at insertion. Their required apsidal rotations of approximately 85 degrees raise the insertion ΔV to almost 1.9 km/sec for an April 4, 1974, arrival date. This exceeds the ΔV capability allocated to the orbit insertion propulsion system ($\sim$1.6 km/sec) by a significant amount.

Aside from illumination, a primary factor in the selection of orbit inclination is planet coverage. A high orbit inclination is desirable as it provides coverage over a wide latitude band. Here again the 120-degree retrograde orbit, i.e., a 60-degree inclination, appears to be superior to the other three orbits considered in Table 3.1-3. However, if the latitude band for which the illumination angle is favorable is considered, the 45-degree inclination orbit is superior.

As indicated earlier, the solar occultation constraint may rule out orbits that are favorable on the basis of photoimaging considerations. Figure 3.1-15 indicates the maximum solar occultation encountered during the 180-day orbit mission for four orbit inclinations as a function of orbit period. The maximum possible occultation for a given orbit size also is shown. This maximum occurs when the Sun-Mars line is in the plane of the orbit, and the Sun is along the line of apsides toward apoapsis.

3.1.3.3 Orbit Insertion Considerations

To enter into an elliptic orbit around Mars, an insertion maneuver is required. This insertion maneuver provides the required ΔV to establish an orbit and reorients periapsis to achieve the illumination desired for imaging. The angle through which periapsis is reoriented is termed apsidal rotation (ϕ). Figure 3.1-16 illustrates the orbit insertion geometry. The apsidal line of the orbit is selected to satisfy mission and system requirements, e.g., photoimaging. The approach asymptote, S, depends on the selected heliocentric transfer trajectory. The position of the approach asymptote relative to Mars can be adjusted, but its direction relative to Mars is fixed.

For a specified approach asymptote, there are two locations for insertion into the elliptical orbit about Mars. The two locations will require different velocity increments (ΔV). As the aiming point is moved further away from Mars, the ΔV requirements approach the same value, i.e., the two locations coalesce. This limiting case occurs when the approach hyperbola and the elliptical orbit are

Table 3.1-3: ORBIT ILLUMINATION CHARACTERISTICS

INCLINATION (Deg)	PERIOD (Hr)	TIME IN ORBIT (Days)	AT 15° ILLUMINATION		AT PERIAPSIS	
			ALTITUDE (km)	LATITUDE (Deg)	ILLUMINATION (Deg)	LATITUDE (Deg)
-30	8.0	0	500	-14.3	15.0	-14.3
		30	520	9.8	22.1	5.2
		60	544	26.8	27.6	22.6
		90	524	29.4	25.4	30.0
		120	506	20.7	20.0	22.5
		150	500	4.6	16.1	5.1
		180	502	-13.0	12.1	-14.3
-45	5.0	0	500	-23.8	15.0	-23.5
		30	835	32.8	42.9	5.7
		60	905	44.7	60.8	33.0
		90	924	27.3	57.2	44.9
		120	1137	- 8.3	66.5	30.0
		150	1102	-34.2	67.6	2.0
		180	895	-44.8	37.3	-26.7
+45	8.5	0	500	9.1	15.0	8.8
		30	548	-15.4	28.5	- 5.3
		60	583	-30.9	34.4	-19.1
		90	570	-39.9	31.0	-31.5
		120	545	-44.4	24.0	-40.9
		150	597	-42.2	20.6	-45.0
		180	708	-43.7	24.6	-42.3
-120	6.5	0	500	16.0	15.0	16.5
		30	501	21.8	13.6	20.6
		60	501	26.3	13.0	24.6
		90	501	30.2	13.1	28.6
		120	501	33.7	13.6	32.5
		150	500	36.7	14.1	36.3
		180	500	40.4	14.3	40.0

- LAUNCH/ARRIVAL DATE 8-25-73/4-4-74
- PERIAPSIS ALTITUDE 500 KM

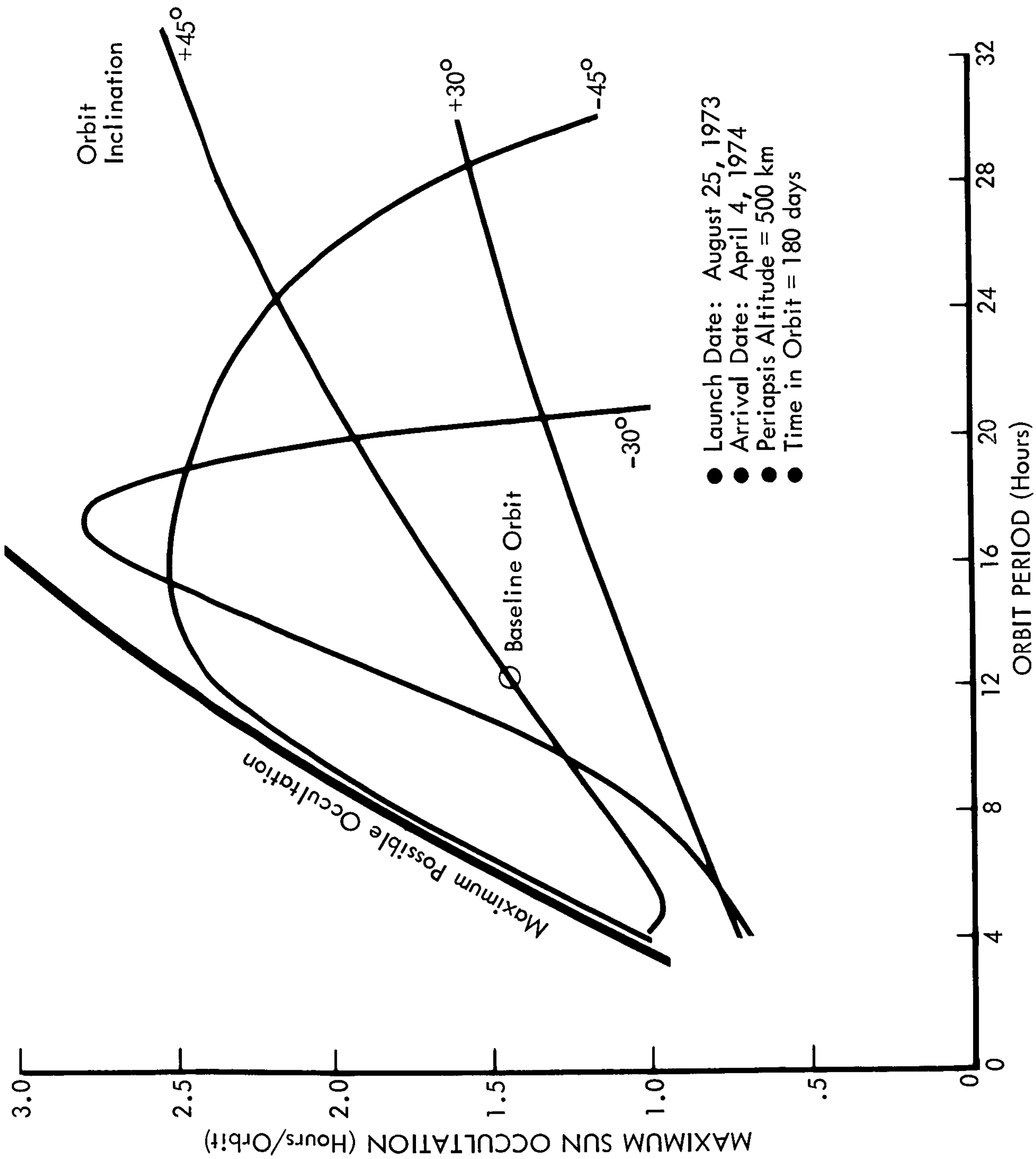


Figure 3.1-15: MAXIMUM SUN OCCULTATION

$\vec{S}$	Unit vector along approach asymptote	R_{P_h}	Periapsis radius of approach hyperbola
$\vec{B}$	Aiming point vector (measure from center of Mars to approach asymptote, in plane normal to asymptote)	R_{P_e}	Periapsis radius of elliptic orbit about Mars
		ϕ	Apsidal rotation (measured from hyperbola periapsis to ellipse periapsis)

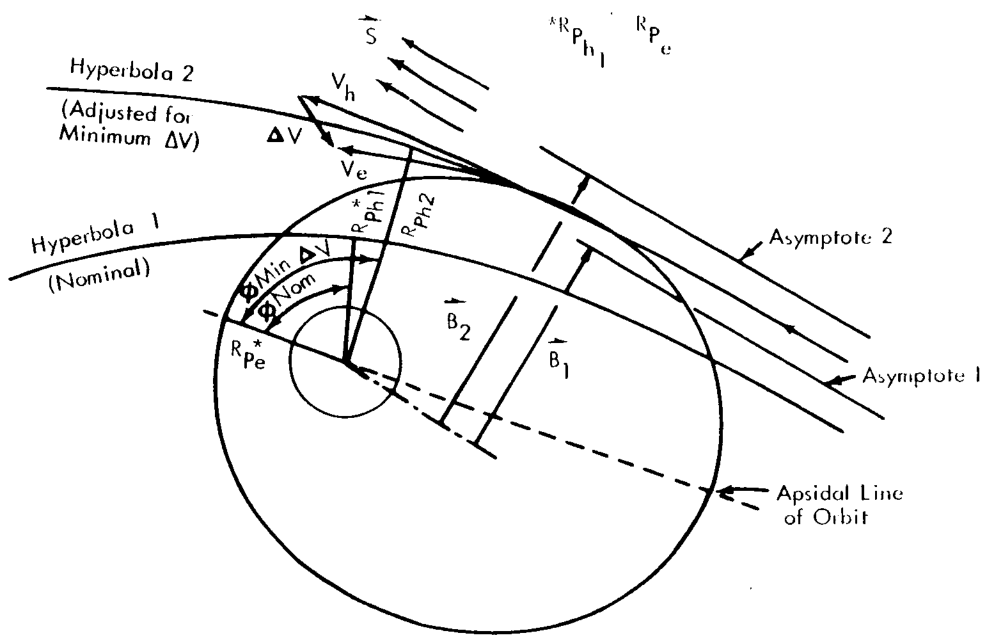


Figure 3.1-16: ORBIT INSERTION GEOMETRY

tangent. The ΔV requirements for inserting into orbits of interest are shown in Figure 3.1-17. Velocity increments are high for low period orbits and orbits that require high apsidal rotation angles. Minimum ΔV requirements occur when no apsidal rotations are required, i.e., insertion occurs at periapsis (of both the hyperbola and the ellipse). The hyperbolic velocity associated with the data in Figure 3.1-17 (V_{HP} = 2.5 km/sec) is typical for arrivals in early April of 1974.

Apsidal rotation angle requirements to place the periapsis of the baseline Mars orbit 15 degrees above the evening terminator are shown in Figure 3.1-18. Apsidal rotation requirements vary from -10 to +40 degrees over the trajectory design envelope for the 1973 mission. These apsidal rotation angles and the attendant approach velocities can be translated into the orbit insertion ΔV requirements indicated by Figure 3.1-19. The data are for the baseline orbit of 500 km periapsis altitude and 18,500 km apoapsis altitude. As can be inferred from Figure 3.1-19, the allocated orbit insertion ΔV of 1.59 km/sec is adequate for attaining the baseline orbit for all arrival-date/launch-date combinations within the trajectory design envelope.

3.1.3.4 Baseline Orbit Assessment

An assessment of the baseline orbit, which was selected as a guide for the spacecraft definition effort, led to the following conclusions and recommendations.

1) The baseline orbit satisfies the planetary quarantine requirement.

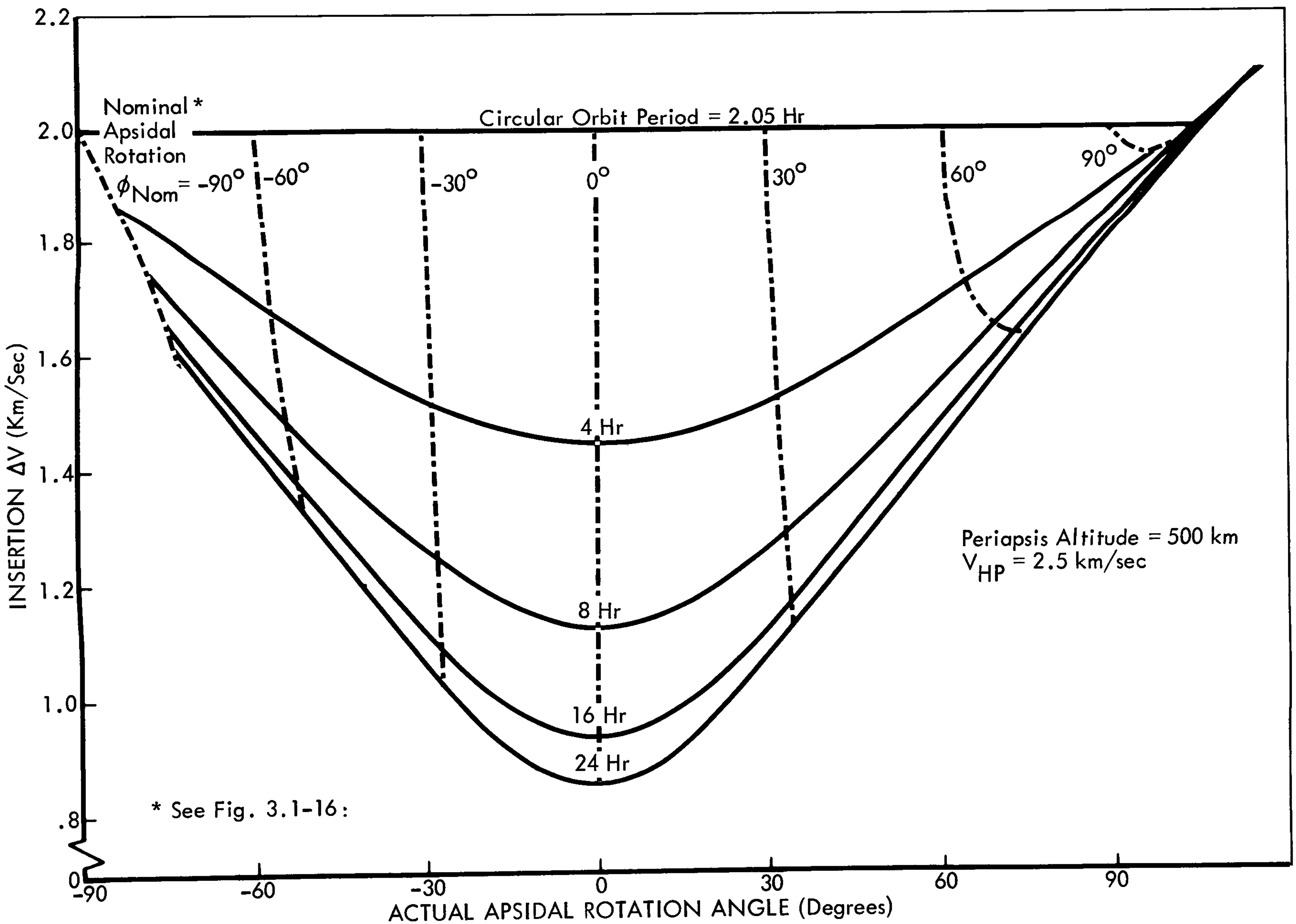


Figure 3.1-17: VELOCITY INCREMENT FOR INSERTION INTO MARS ORBIT AS FUNCTION OF APSIDAL ROTATION

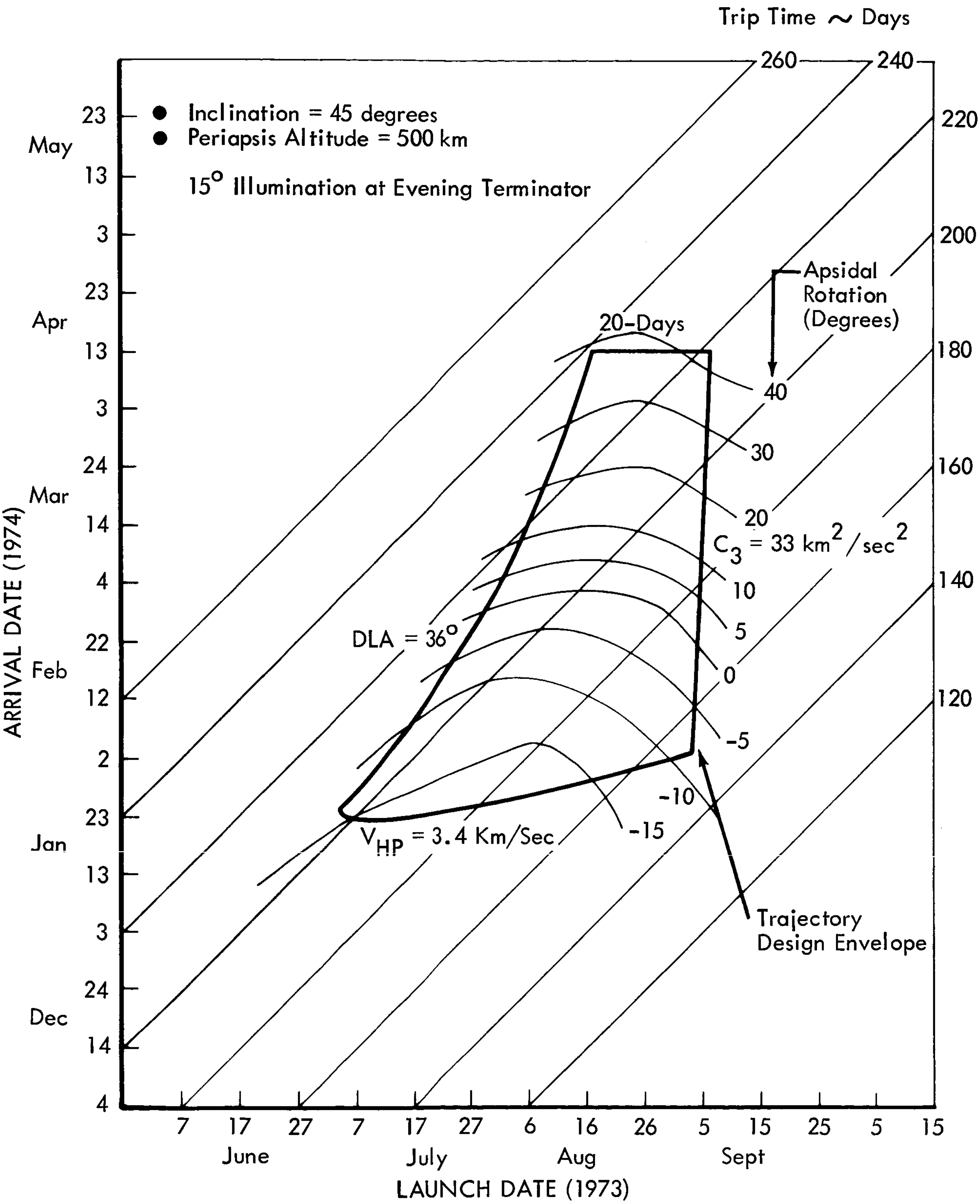


Figure 3.1-18 : REQUIRED APSIDAL ROTATION

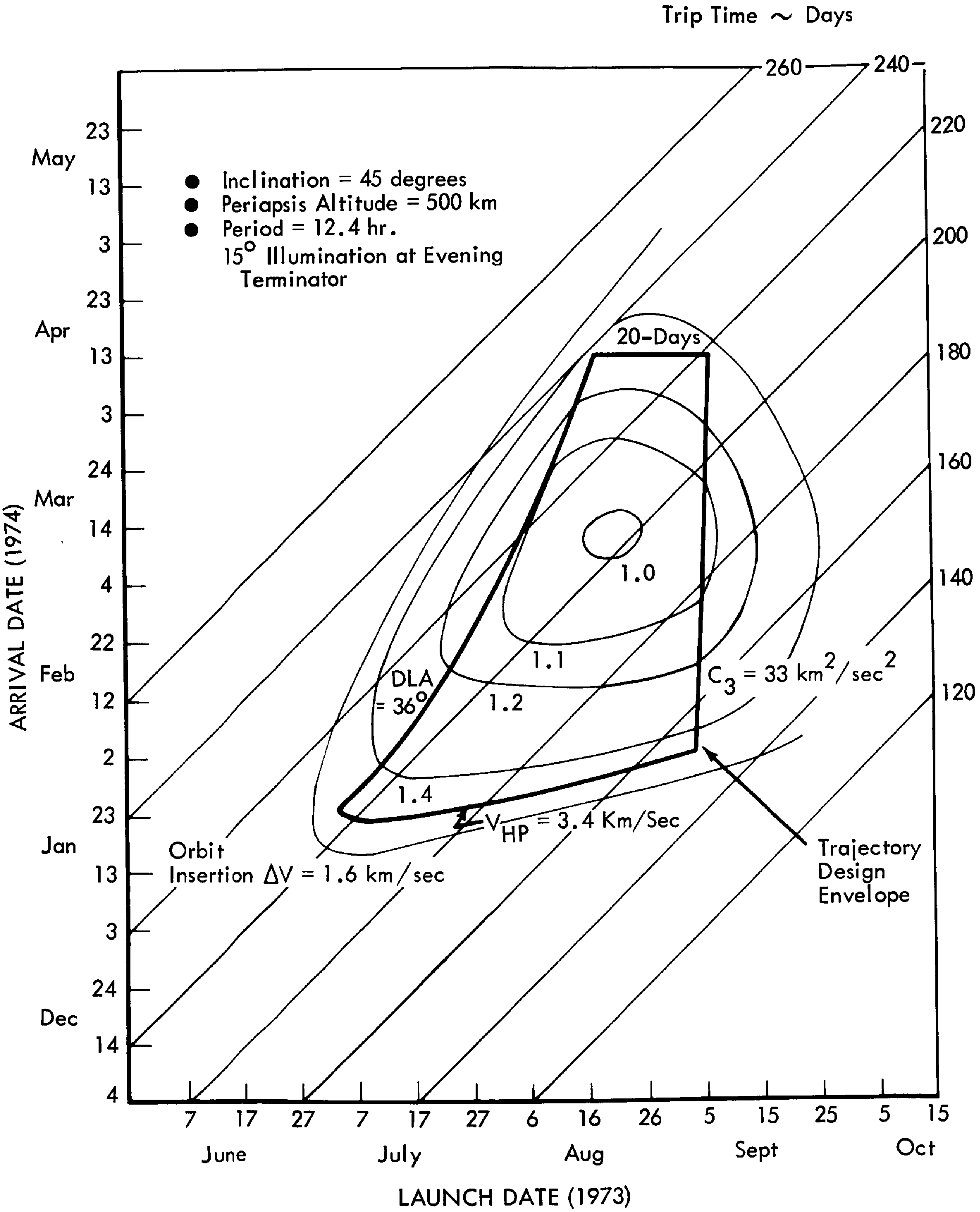


Figure 3.1-19: ORBIT INSERTION ΔV

2) The baseline orbit satisfies the Sun occultation constraint.

3) The baseline orbit satisfies the Earth occultation experiment requirements in the early and middle phases of the orbital mission.

4) The baseline orbit provides adequate latitude coverage, wave-of-darkening coverage and periapsis or near-periapsis illumination, for conducting the photoimaging experiment.

5) The allocated orbit insertion ΔV of 1.59 km/sec is adequate for inserting into the baseline orbit, including apsidal rotation for 15-degree illumination at periapsis, for all launch-date/arrival-date combinations within the trajectory design envelope.

6) The orbit trim ΔV allocation of 150 meters/sec is adequate for March/April arrival dates. An additional trim ΔV allocation of 60 m/sec is required to accommodate early (February) arrival dates.

7) Alternate orbits that might better satisfy the photoimaging experiment objectives should be examined.

3.1.4 Earth-Mars Trajectories - 1973

For each of the Earth-Mars mission opportunities (roughly every 25 months), there exists a family of Earth-to-Mars trajectories. Each launch-date/arrival-date combination defines such a trajectory. Trajectory parameters for the 1973 mission are plotted on Figure 3.1-20. The trajectories are divided into two categories, designated Type I and Type II, according to whether the heliocentric trajectory subtends a central angle at the Sun of less or more than 180 degrees. Because Type I trajectories (subtended central angles less than 180 degrees) have shorter trip times and communication distances, Type I trajectories are preferred for the 1973 mission design. Their launch and insertion energy requirements can be met with current weight allocations.

To assess the capability of the system to perform a mission, its weight, propulsion capability, launch period, and launch range restrictions must be considered. Table 3.1-4 presents the weight allocations for the two 1973 mission cases.

The first important constraint to consider in evaluating a transfer trajectory is booster capability. Figure 3.1-21 presents the Saturn V payload capability versus the geocentric launch energy, C_3. As indicated, the highest C_3 that the Saturn V booster can provide for the Case A and Case B system weights is 33 and 28.8 km^2/sec^2, respectively.

The spacecraft's propulsion subsystem capability sets another constraint on interplanetary transfer trajectory selection. The highest hyperbolic excess velocity (V_{HP}) that can be accommodated is directly related to the ΔV allocation of the spacecraft. A total ΔV capability of 1950 m/sec was allocated to each planetary vehicle. Deducting 210 m/sec for midcourse and arrival time biasing maneuvers, and 150 m/sec for orbit trim, leaves a ΔV of 1590 m/sec available for insertion. To insert the planetary vehicle into the design orbit with an apsidal rotation of ±20 degrees (a conservative assumption for early arrival), the maximum value of V_{HP} that can be accommodated is 3.4 km/sec (a 40 m/sec reserve ΔV was assumed).

Table 3.1-4 : 1973 VOYAGER SYSTEM WEIGHT ALLOCATION (pounds)

	CASE A		CASE B	
Total Planetary Vehicle		20750		22750
Total Dry Spacecraft	5650		5800	
Propulsion subsystem inerts	(2050)		(2100)	
Spacecraft	(2940)		(3030)	
Science	(390)		(390)	
Contingency	(270)		(280)	
Usable Propellants	10100		10950	
Capsule	5000		6000	
Planetary Vehicle Adapter Allowance		250		250
Total per Planetary Vehicle		21000		23000
Mission Contingency		5000		5000
Total Net Payload per Launch		47000		51000

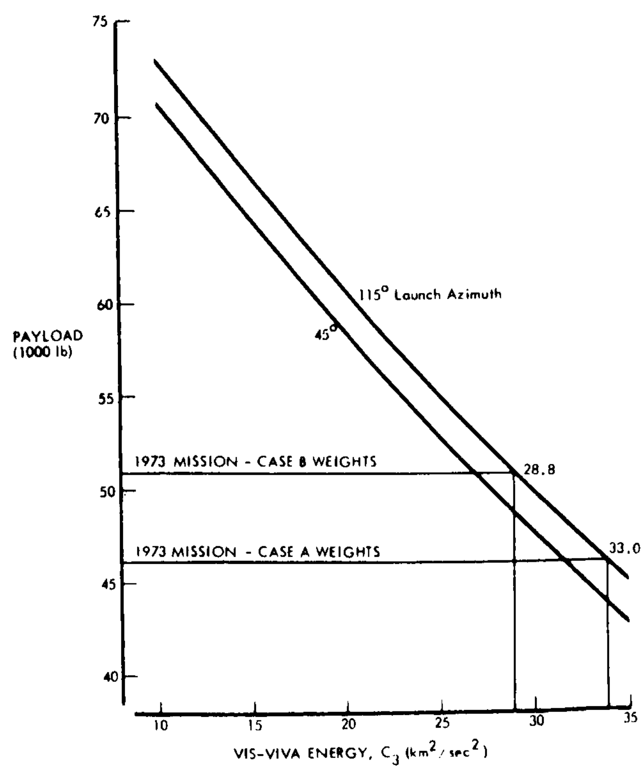


Figure 3.1-21: VOYAGER PAYLOAD VS. C_3 FOR SATURN V

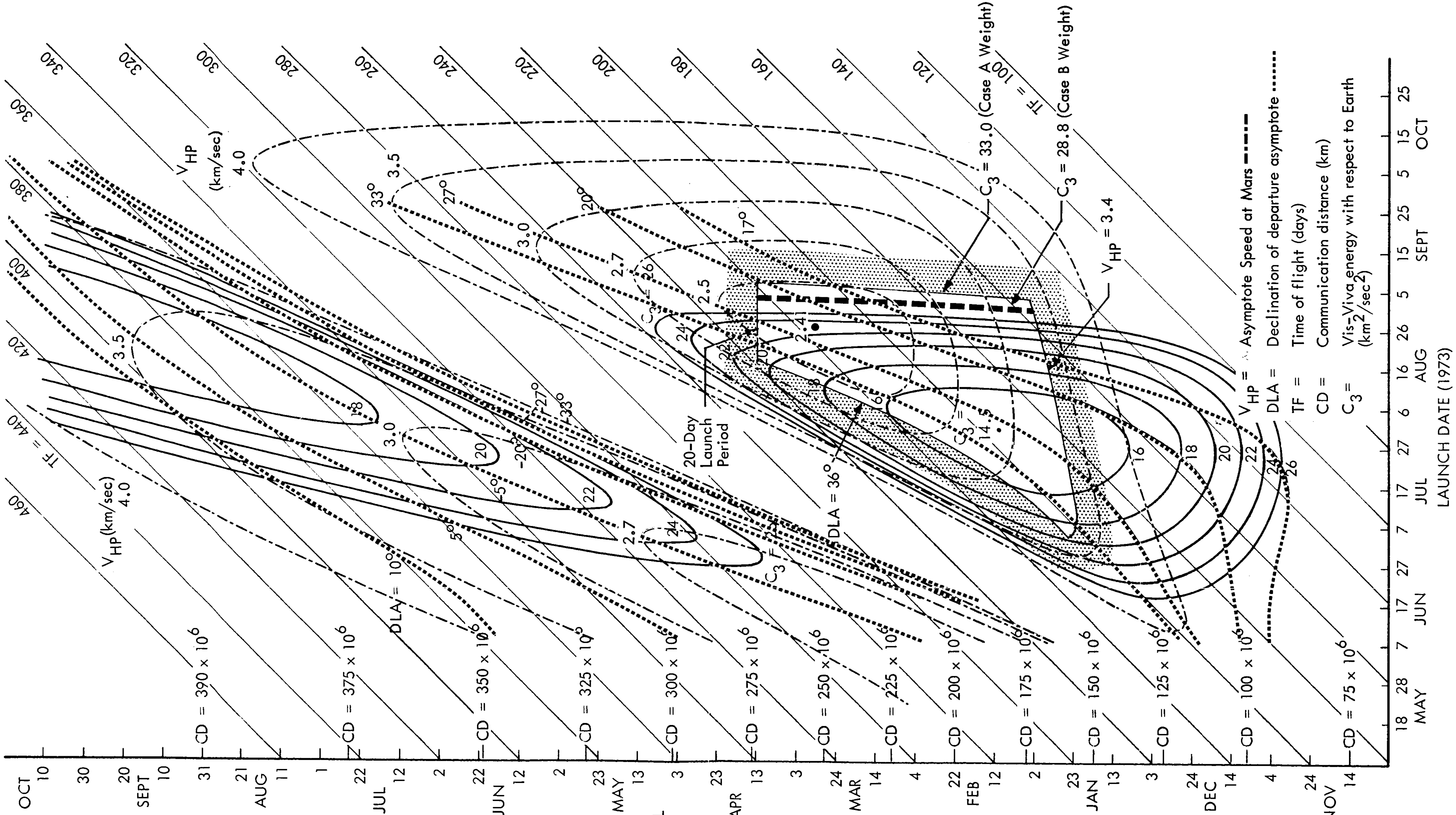


Figure 3.1-20: EARTH-MARS TRANSFER CHARACTERISTICS — 1973 TYPES I AND II

The launch azimuth restrictions of the Eastern Test Range (ETR) impose an additional constraint. Figure 3.1-22 shows the relationship of launch azimuth, launch window, and declination of the launch asymptote (DLA). Launch azimuth values between 90 and 115 degrees were specified for the 1973 launch opportunity. To provide a 2-hour per day launch window, it is necessary to provide DLA's (departure) below 36 degrees.

The system capability envelope (see Figure 3.1-20) is completed by the latest arrival date constraint which results from the requirement of a minimum reasonable launch period of 20 days.

The combination of the above capability constraints allows the construction of the 1973 mission trajectory envelope bounded by a C_3 of 33 km^2/sec^2 on the right (Case A weights), a DLA limit of 36 degrees on the left, a V_{HP} value of 3.4 km/sec on the bottom, and a 20-day launch period on the top. This envelope encompasses the launch date, arrival date, and energy capabilities of the Voyager system. The heavy dashed line on the right side of the envelope represents the capability limit (C_3 = 28.8 km^2/sec^2) for Case B weights. For the latter case, the available launch period for the April 13 arrival date is reduced to 16 days. Thus, for Case B, the latest arrival date allowed is April 2. Figure 3.1-23 presents detailed 1973 interplanetary trajectory energy parameters within the capability envelope.

3.1.4.1 Aiming Point Selection Criteria

By combining the features of the baseline orbit with the trajectory capability envelope, aim point selection can be made. The approach aiming coordinates are defined in Figure 3.1-24. The position of Sun and Earth with respect to the

Definitions:

S	S-Vector is the direction of the approach hyperbola asymptote at Mars arrival.
T	Orthogonal to S, and lying in the ecliptic plane.
R	Forms a right handed orthogonal system with S, T.
ZAP	Angle between S and vector to Sun
ZAE	Angle between S and vector to Earth.
ETS	Angle measured in R-T plane from T to the negative projection of the Sun vector.
ETE	Angle measured in R-T plane from T to the negative projection of the Earth vector.

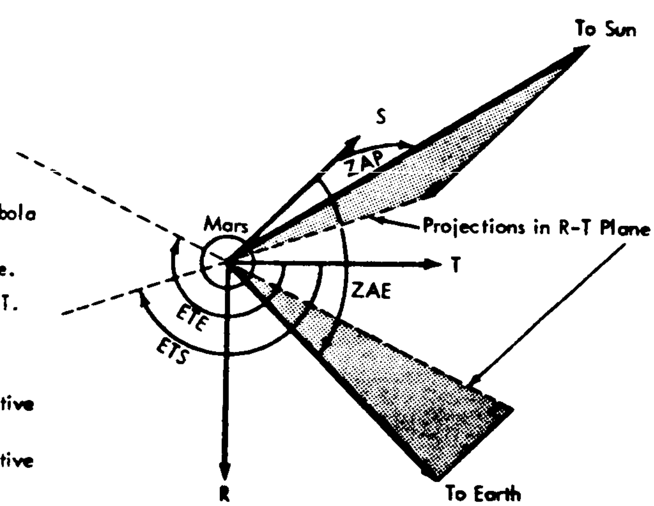


Figure 3.1-24: SUN-EARTH-SPACECRAFT GEOMETRY AT MARS ARRIVAL

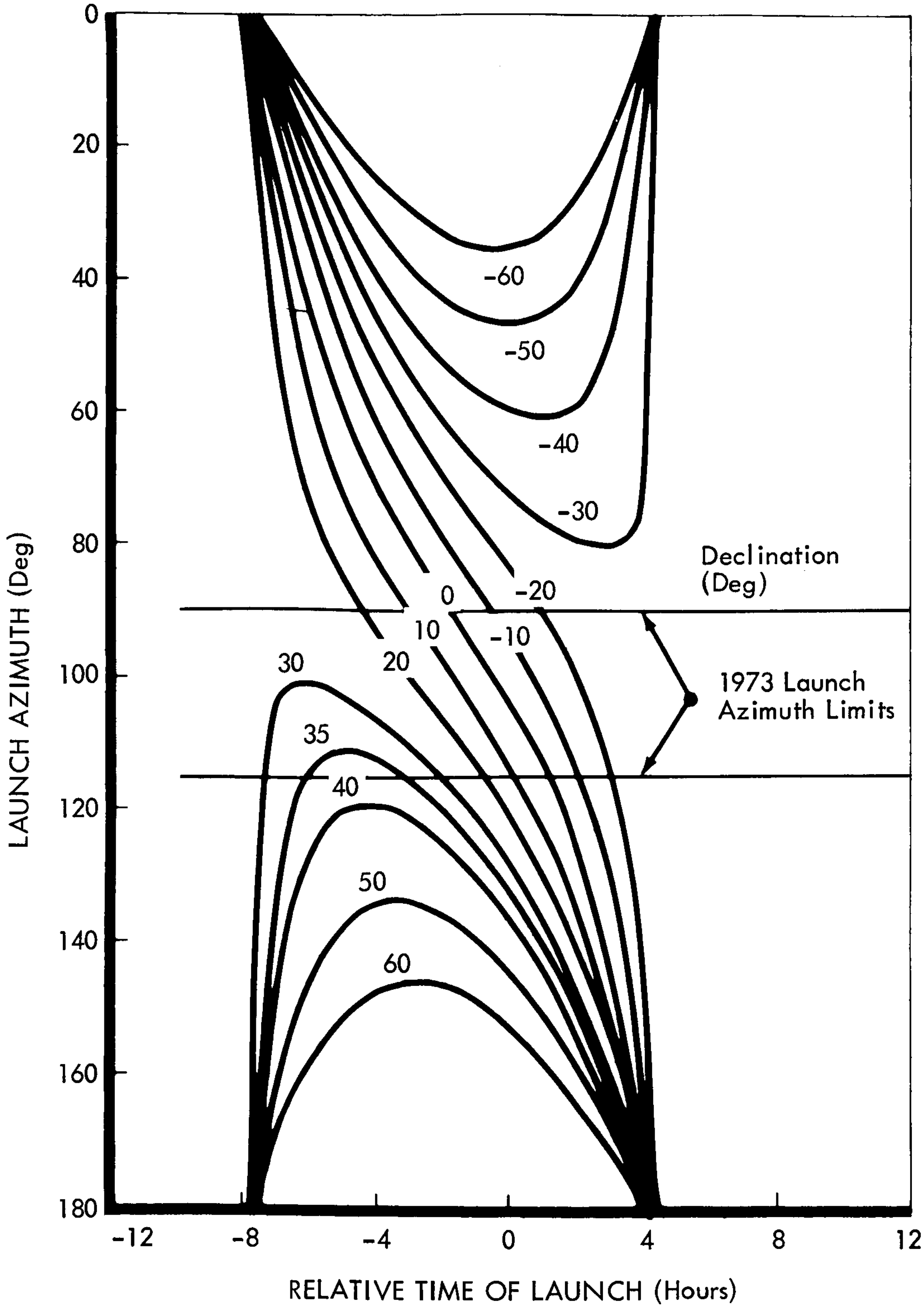


Figure 3.1-22: AVAILABLE LAUNCH WINDOWS

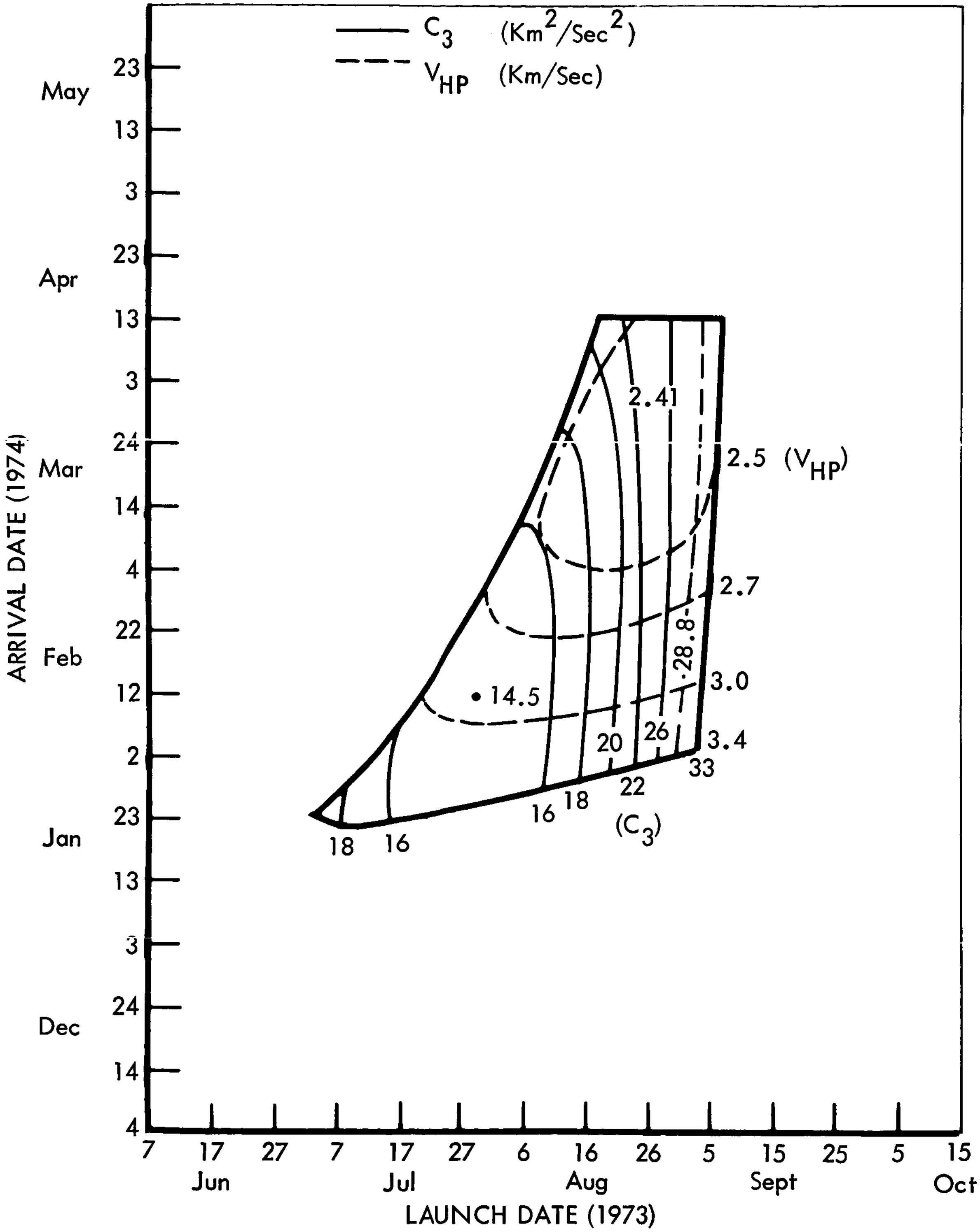


Figure 3.1-23: OPERATIONAL ENVELOPE FOR 1973 TYPE I TRAJECTORY ANALYSIS — Energy and Velocity Data

approach-aiming-coordinate system is indicated. This coordinate system is based on the direction of the approach hyperbolic asymptote, given by S, and the T axis. T is normal to S and parallel to the ecliptic plane. The third coordinate axis, R, is determined from R x S = T. An aiming point is the point at which the asymptote of the approach hyperbola passes through the RT plane. The Sun angular coordinates, ZAP and ETS, are plotted in Figure 3.1-25 for the launch and arrival dates of interest. (See Figure 3.1-24 for definitions.) Similar coordinates for the Earth, ZAE and ETE, are shown in Figure 3.1-26.

Figure 3.1-25 indicates that all direct orbits will be free from solar occultation during the insertion phase. In addition, the ZAP angles are such that periapsis of the approach hyperbola will occur near the evening terminator. This will result in low ΔV's to achieve a Mars orbit, and good illumination for topographic imagery. Figure 3.1-26 shows that Earth occultation geometry is more favorable for the later arrival dates and direct orbits. When the ZAE angle approaches 180 degrees, as is the case for early arrival trajectories, occultation at insertion is likely for many aiming points. In such a case, detailed occultation plots, such as shown in Figures 3.1-27, 3.1-28, and 3.1-29, are required. These figures show the available sector of B vectors (aiming vector in the RT plane from the center of Mars to the approach asymptote) that do not result in Earth occultation at orbit insertion. The available B vector sector grows steadily larger the later the arrival date. The size of the available sector is directly translatable into orbit inclinations as indicated in Figures 3.1-27, 3.1-28, and 3.1-29.

Figure 3.1-30 indicates the latitude of the vertical impact (LVI) point, the Mars latitude of impact for an approach trajectory aimed at the center of Mars. The inclination of the desired Mars orbit must be greater than the latitude of the vertical impact point if a plane change maneuver at insertion is to be avoided. All launch and arrival date combinations during the 1973 opportunity have low LVI values. Consequently, the LVI does not influence the selection of launch and arrival dates in 1973.

3.1.4.2 Selection of Aiming Point - 1973

Aim points for the trans-Mars trajectory are selected:

1) To maximize the probability that the trans-Mars trajectory will have the required encounter conditions.

2) To maintain the probability of contamination of Mars below the allocation.

Because of the contamination (planetary quarantine) constraint, and navigation and control errors, the aim points for injection from Earth orbit and for each midcourse correction are different, and require separate solutions. The allocated probability of contaminating Mars by accidental planetary vehicle impact is 1×10^{-5} or less. To satisfy this allocation, aiming points must be selected so that the probability of spacecraft impact from trans-Mars trajectory will be less than 2×10^{-6} (see Figure 3.1-2).

The selection of the aiming points for the 1973 mission is discussed below. The nomenclature for aiming point analysis is indicated in Figure 3.1-31.

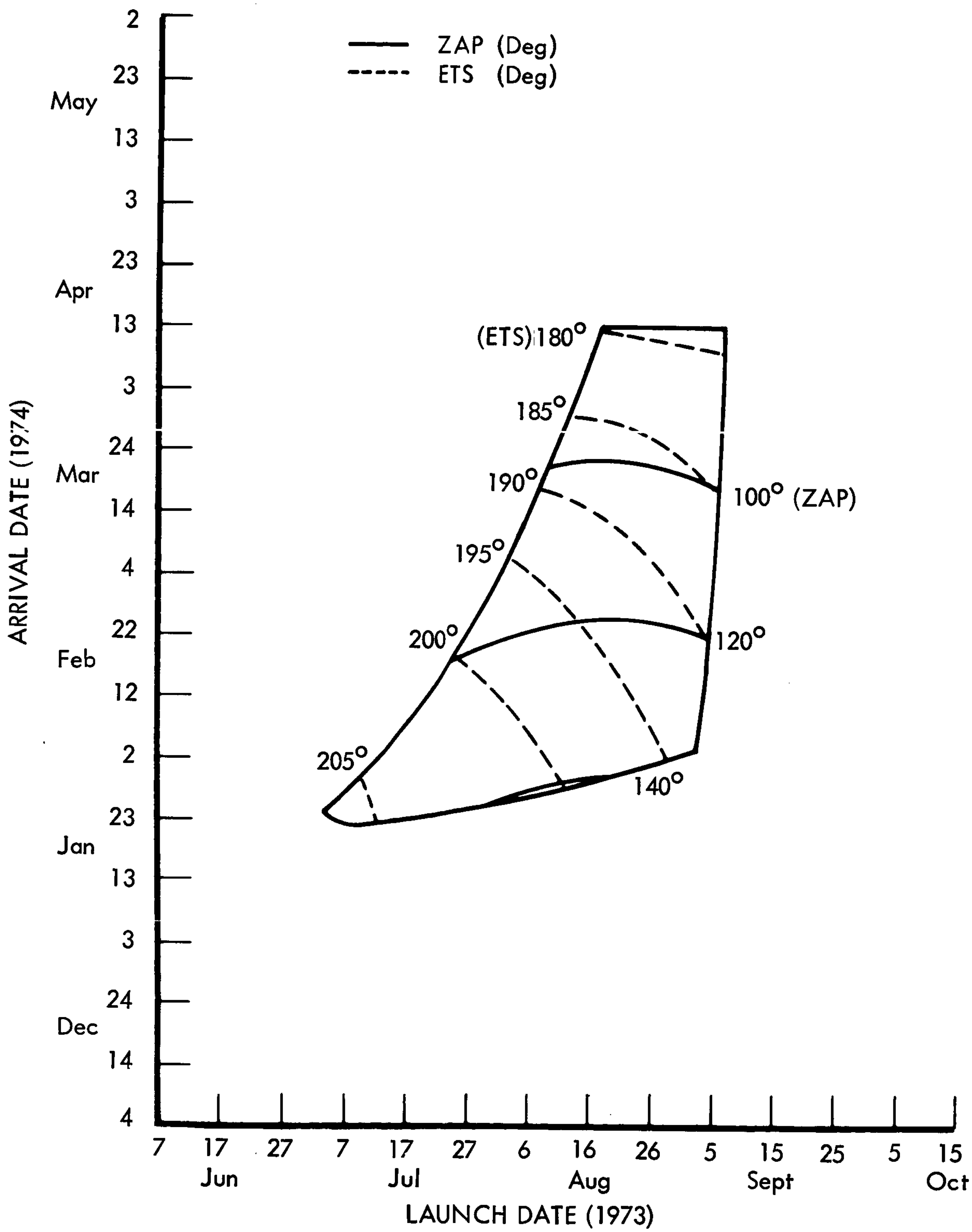


Figure 3.1-25: OPERATIONAL ENVELOPE FOR 1973 TYPE I TRAJECTORIES — SUN ANGLE DATA

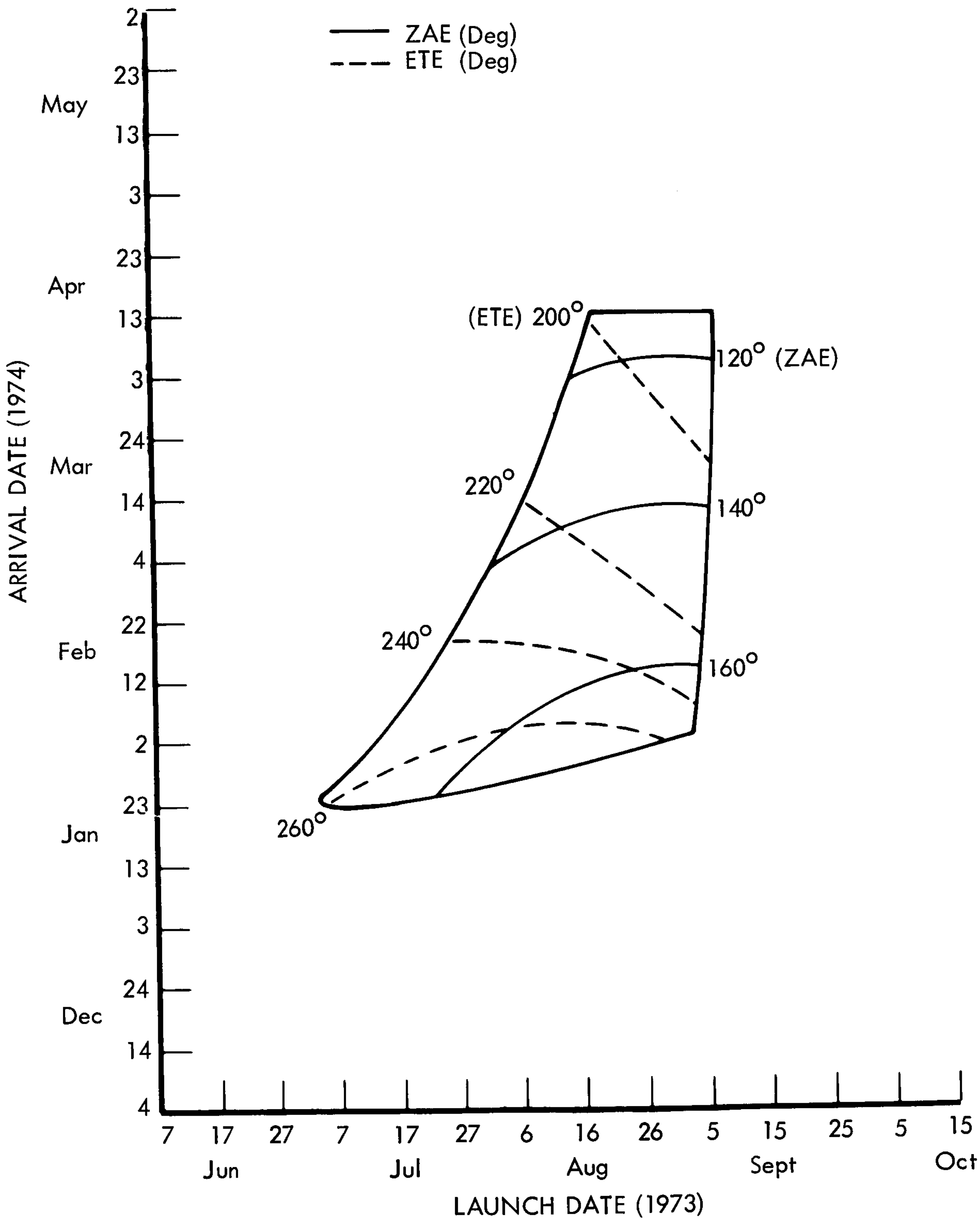


Figure 3.1-26: OPERATIONAL ENVELOPE FOR 1973 TYPE I TRAJECTORIES — EARTH ANGLE DATA

- LAUNCH AUGUST 25, 1973
- ARRIVAL $V_{HP} = 2.41$ km/sec

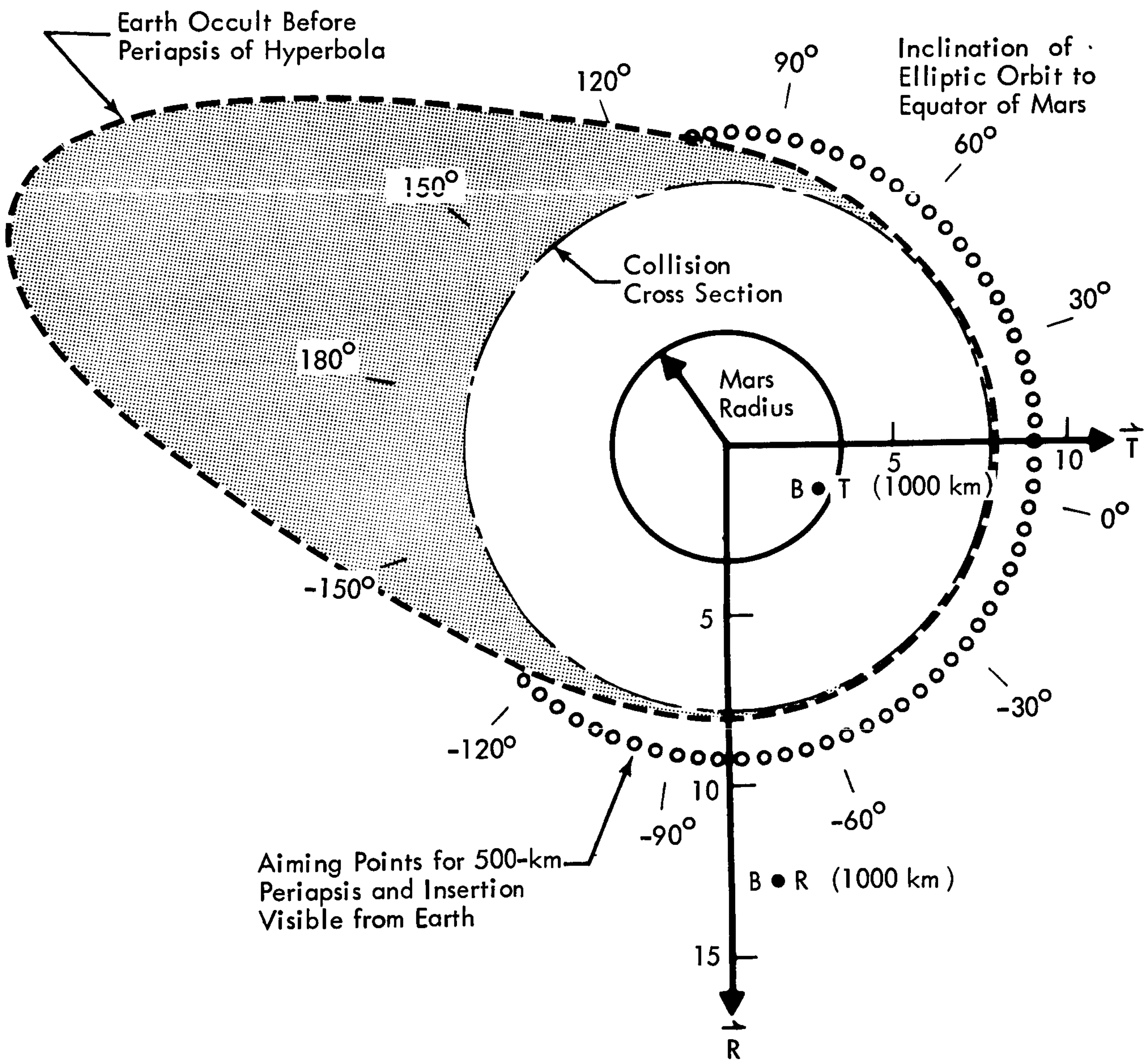


Figure 3.1-29: EARTH OCCULTATION AT ARRIVAL, APRIL 4, 1974
LATE ARRIVAL

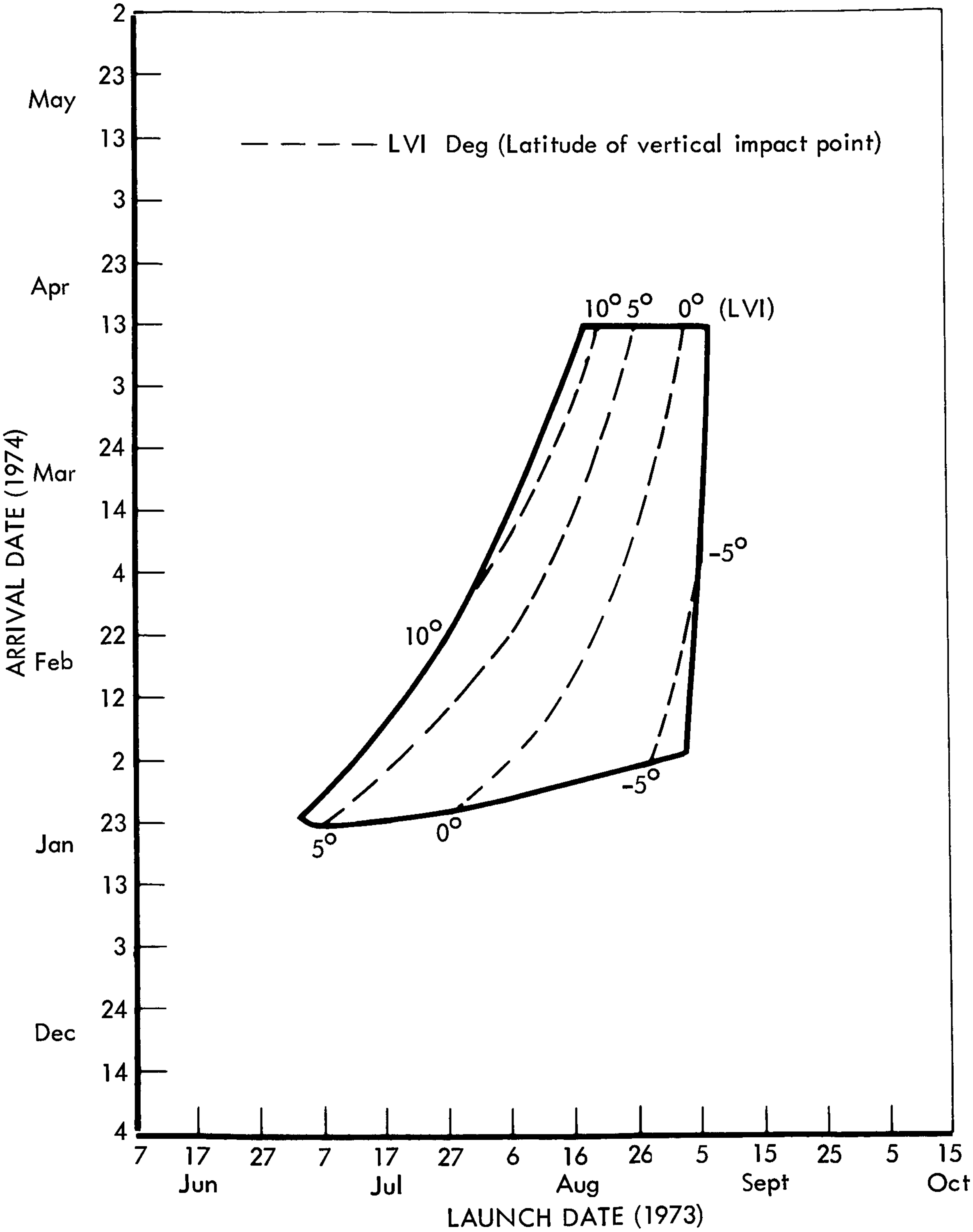
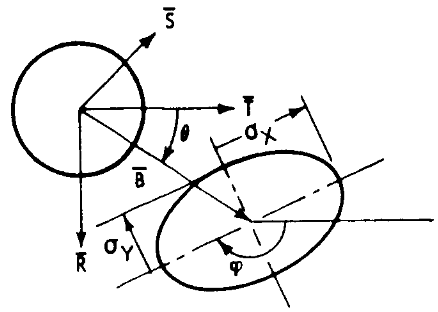


Figure 3.1- 30: OPERATIONAL ENVELOPE FOR 1973 TYPE I TRAJECTORY ANALYSIS - LVI DATA



$\overline{R}$, $\overline{S}$, $\overline{T}$ Coordinate System:

$\overline{S}$ - Direction of asymptote of hyperbolic trajectory, approaching Mars.

$\overline{T}$ - Normal to $\overline{S}$ and in the plane of the ecliptic.

$\overline{R}$ - Directed so that $\overline{R}$, $\overline{S}$, $\overline{T}$ is a right-hand triad.

$\overline{B}$ - A vector in the R, T plane directed from the planet center normal to the approach asymptote.

B - Magnitude of $\overline{B}$

θ - Direction of $\overline{B}$

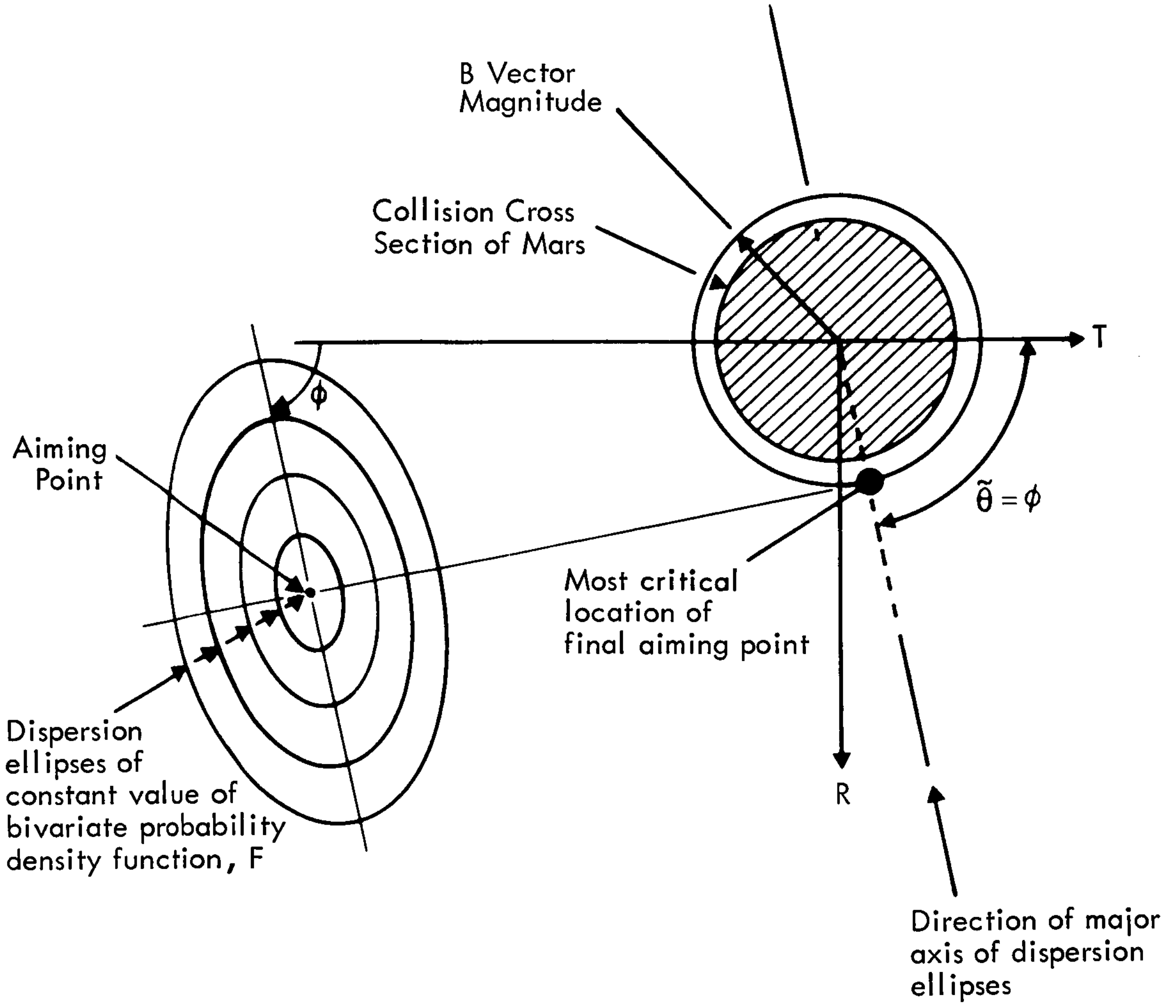
σ x, σ y - Standard deviations of the dispersion of B.

φ - Orientation of dispersion of B.

Figure 3.1-31: NOMENCLATURE FOR AIMING POINT ANALYSIS

<u>Final Aim Point Selection</u> -- After a particular launch and arrival date combination has been selected, the S vector is uniquely defined at Mars. It remains to pick a B vector in order to define the desired final aiming point. The selection of a B vector is determined (for coplanar insertion) by the intended baseline orbit, and the desire to minimize insertion ΔV. The intended elliptic orbit periapsis at 500-km altitude establishes a lower limit for the periapsis altitude of the approach hyperbolic path. It is necessary to verify that this limit is acceptable from planetary quarantine consideration. This is accomplished as follows: First, the resultant magnitude of the B vector is calculated; the most adverse value of the direction of the B vector is then selected, and the trajectory aimed at that point; the Mars impact probability for this aim point must be shown to be less than 2×10^{-6} for that B vector to be acceptable.

The magnitude of the B vector is computed by taking the minimum allowable periapsis altitude of 500 km and combining it with the hyperbolic approach speed for the selected launch-date arrival-date combination. A value of the aim angle θ is now selected (see Figure 3.1-31) so that the B vector is located in the most adverse geometric relation to the arrival dispersion ellipses resulting from injection into the trans-Mars orbit. The final aiming point may be placed anywhere in the RT plane, subject to the two constraints expressed as definite integrals in Figure 3.1-32. The small black circle in Figure 3.1-32 represents the area on the RT plane that encloses all Mars encounter trajectories that the spacecraft guidance and control subsystem can convert into acceptable elliptic orbits around Mars; the black circle, therefore, represents the correct encounter conditions. The first integral relation on Figure 3.1-32 maximizes the probability of correct encounter conditions. The second integral relation ensures that the contamination probability allocation is met.



- Required: $\int_{\bullet} F\,dA$ to be a maximum
- Required: $\int_{\oslash} F\,dA$ to be $\leq 2 \times 10^{-6}$

Figure 3.1-32: SELECTION OF MOST CRITICAL LOCATION OF FINAL AIM POINT

The geometry of Figure 3.1-32 indicates that the most critical value of the orientation $\tilde{\theta}$ of the B vector occurs when $\tilde{\theta} = \phi$, where ϕ is the angle between the major axis of the dispersion ellipse and the T axis. This geometry is most critical because it maximizes the shift required to move the initial aiming point to the final aim point on the RT plane.

Following the above procedure, the specific final aiming points selected for the baseline orbit are shown in Table 3.1-5.

Table 3.1-5: Baseline Orbit Final Aim Points

	Early Trajectory Arrival Date	Middle Trajectory Arrival Date	Late Trajectory Arrival Date
Launch date	8-4-73	8-17-73	8-25-73
Arrival date	2-4-74	3-1-74	4-4-74
Radius (km) of collision cross section	6,600	7,500	8,000
Location of final aim point B (km)	7090	8160	9160
B.T (km)	5,773	2,904	1,839
B.R (km)	-4,133	-7,605	8,960
θ (deg)	-35.6	-69.1	78.4

Initial Aiming Point Selection -- The initial aiming point is that used for injecting from Earth parking orbit into the trans-Mars trajectory. Following shutdown of the SIVB stage second burn, the following objects are all traveling toward Mars along nearly identical ballistic trajectories: Two planetary vehicles; the spent SIVB stage; the cylindrical section of the aft shroud; and minor particles ejected during the separation maneuver. All these must be considered in selecting the initial aim point. The aim point selection problem will be solved first for a single spacecraft. Then the effects of the additional traveling bodies will be examined.

The selection of the initial aim point requires a representative convariance matrix of shutdown errors for the SIVB stage of the Saturn V. This matrix is rotated into the proper coordinates for each nominal interplanetary trajectory, and mapped to Mars encounter. This process results in the encounter dispersion parameters in Table 3.1-6.

Table 3.1-6: Errors at Mars Encounter

	Early Arrival Date	Middle Arrival Date	Late Arrival Date
σ x (km)	281,500	220,000	269,000
σ y (km)	4,920	7,260	9,020
ϕ (deg)	-35.6	-69.1	78.4

By following the rationale indicated in Figure 3.1-32, and solving the integral equation $\int_0 FdA = 2 \times 10^{-6}$, the results shown in Figure 3.1-33 are obtained.

LOCATION OF INITIAL AIM POINT

	E EARLY ARRIVAL (Feb)	M MID ARRIVAL (Mar)	L LATE ARRIVAL (Apr)
B (Km)	25,000	35,000	41,700
$\bar{B} \cdot \bar{T}$ (Km)	+19,700	+ 34,700	- 38,100
$\bar{B} \cdot \bar{R}$ (Km)	+15,300	+ 4,500	+17,100

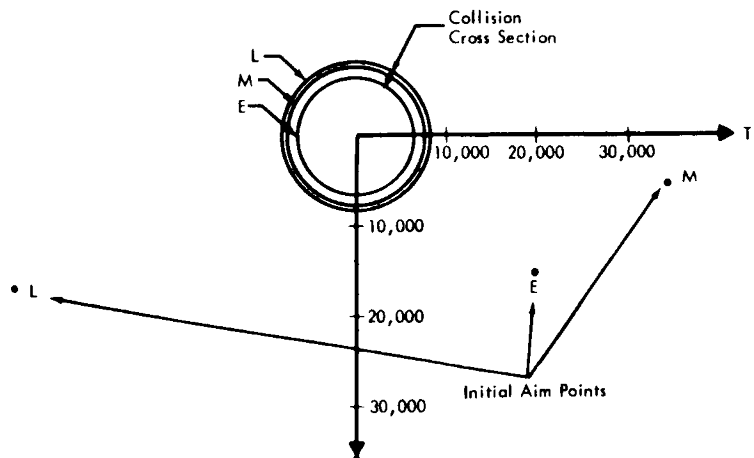


Figure 3.1-33: SELECTION OF INITIAL AIM POINT

Aim Point Selection for First Midcourse Correction -- The purpose of the first midcourse trajectory correction is:

1) To separate Mars encounter arrival time by 4 days for each of the two planetary vehicles (i.e., 4 days for one and -4 days for the other).

2) To correct random errors in B.T and B.R.

3) To shift the aiming point in the RT plane closer to Mars because error ellipses are now smaller and less eccentric.

The following assumptions were made in selecting the aim point for the first midcourse:

1) The probability of success of a second midcourse maneuver is 0.994.

2) The aim point must be located to satisfy a probability of impact of 33.3×10^{-4}. (This is a consequence of the first assumption.)

3) The selected spacecraft (conservative) control error set for all trajectory corrections (excluding engine errors) is as follows:

Error	$3\ \sigma$ Value
Pointing	42 milliradians
ΔV Proportional	1.0%
ΔV Resolution	0.02 m/second

4) The nominal times for midcourse corrections are chosen, on the basis of past studies, as follows: first midcourse at launch plus 5 days; second midcourse at launch plus 25 days; and third midcourse at Mars encounter minus 40 days.

The selection of the first midcourse aiming point is accomplished by:

1) Determining navigation errors;

2) Mapping injection error to maneuver time;

3) Computing guidance coefficients;

4) Computing covariance of ΔV, the ΔV for arrival time bias, and nominal aim point shift;

5) Computing error in state due to control errors;

6) Mapping navigational and control errors into encounter parameter, errors;

7) Determining a first midcourse aim point, from the encounter dispersion, that satisfies the quarantine constraint;

8) Iterating the above process until changes in dispersion ellipses and aiming point become acceptably small.

Resulting first midcourse aiming points for the early, middle, and late trajectories are illustrated in Figure 3.1-34.

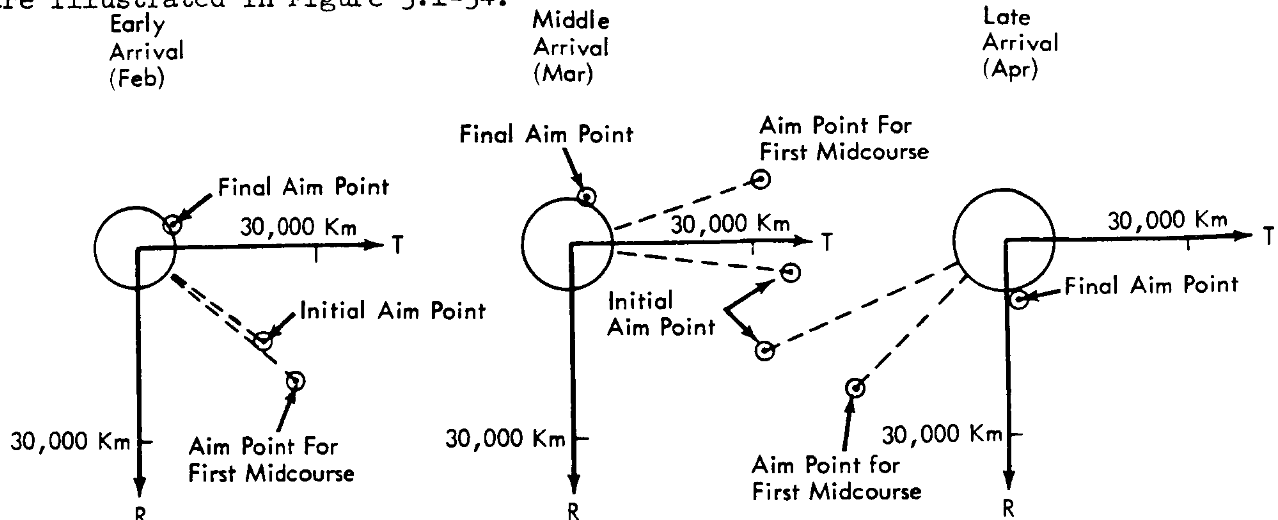


Figure 3.1-34: EFFECTS OF ARRIVAL DATE ON AIM POINT SELECTION

Figure 3.1-34 shows that the length of the B vector to the aiming point for the first midcourse maneuver is nearly the same as the length of the B vector to the initial aiming point. This is caused by the 4-day arrival date separation maneuver which is combined with the first midcourse maneuver. This arrival date biasing requires a ΔV (see Figure 3.1-35) of from 3 to 10 times the size of the ΔV required to null random trajectory errors. As a result, control errors are significant. The selection of better control accuracy than that assumed here will reduce the magnitude of the B vector for the first midcourse aiming point.

The accomplishment of the full 8 days of arrival date biasing by maneuvering only one of the planetary vehicles is not favored because of the large amount of propellant required, and the consequent increase in encounter errors. The ΔV requirement is almost double that shown in Figure 3.1-35.

Aiming Point Selection for Second Midcourse Maneuver -- The purpose of the second trajectory correction maneuver is:

1) To correct random errors in the B vector components in the aim plane, i.e., in in B.T and B.R.

2) To shift the aiming point closer to Mars to capitalize on the reduction in size of the error ellipse.

For selecting the second midcourse maneuver aiming point, the probabilities of success of the third midcourse correction were assumed to be as follows:

Early trajectory	0.964
Middle trajectory	0.961
Late trajectory	0.954

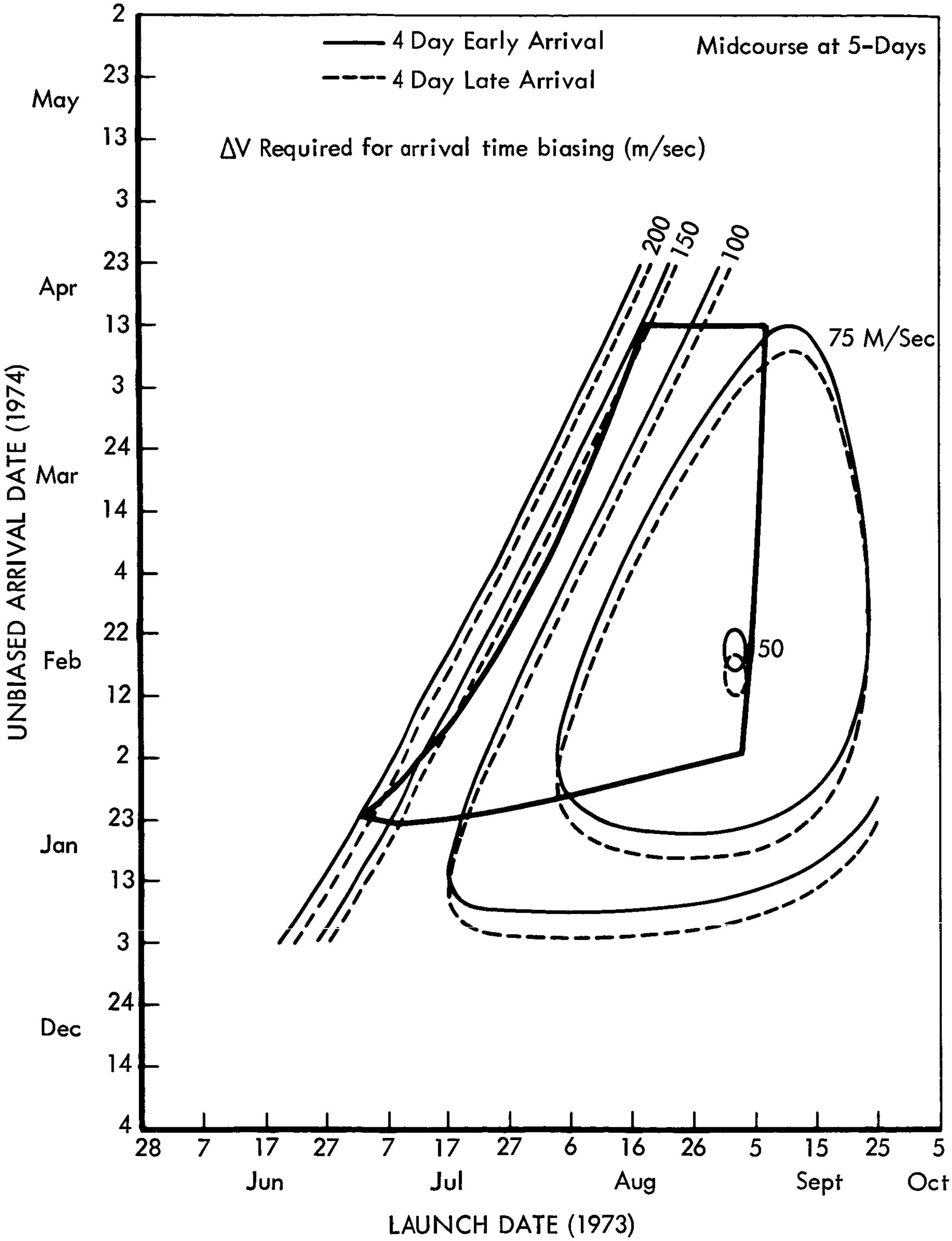


Figure 3.1- 35: IMPLICATIONS OF ARRIVAL-TIME-BIASING ΔV ON ARRIVAL DATE — 4-DAY BIAS

The preceding probabilities reflect the variation in trip times. From these maneuver success probabilities, the aiming points for the second midcourse correction must satisfy the following impact probabilities:

Early trajectory	5.59×10^{-5}
Middle trajectory	5.09×10^{-5}
Late trajectory	4.30×10^{-5}

Aim points were determined in the same manner as described for first midcourse. They are summarized in Table 3.1-7

Table 3.1-7: AIM POINT SUMMARY

LAUNCH DATE	MANEUVER	B.T (km)	B.R (km)	θ (deg)	ϕ (deg)	σx (km)	σy (km)
8/4/73	Injection	19,712	15,337	54.4	-35.6	281,460	4,918
	1st M.C	25,676	21,619	-35.6	-37.7	16,909	8,918
	2nd M.C	7,691	- 5,506	-35.6	-49.7	762	390
	3rd M.C.	5,773	- 4,133	-35.6	-78.9	103	102
8/17/73	Injection	34,734	4,549	20.9	-69.1	220,000	7,263
	1st M.C	31,323	-10,392	-69.1	84.4	17,934	8,364
	2nd M.C	3,563	- 9,332	-69.1	74.5	810	245
	3rd M.C	2,904	- 7,605	-69.1	68.8	104	101
8/25/73	Injection	-38,005	17,149	168.5	78.4	268,826	9,024
	1st M.C	-25,016	24,341	78.4	60.2	45,913	9,637
	2nd M.C	2,767	13,481	78.4	55.3	1,708	267
	3rd M.C	1,839	8,960	78.4	54.2	108	101

Aiming Point Selection for Third Midcourse Maneuver -- The purpose of the third midcourse maneuver is to correct random errors in B.T and B.R., and to shift the aiming point.

The aiming point for the third midcourse correction must satisfy a probability of impact of 2×10^{-6}. Aiming points were determined in the same manner as the previous maneuvers, and are also detailed in Table 3.1-7.

Aiming Point Selection Summary -- Guidance aiming points and dispersion data for the trans-Mars trajectories for the three representative trajectories chosen from the 1973 launch opportunity are given in Table 3.1-7.

By comparing Tables 3.1-7 and 3.1-5, the aiming points for the third mid-course maneuvers are identical with the final aiming points required to attain the baseline orbit. This establishes the fact that the trans-Mars trajectory can be aimed to achieve a hyperbolic periapsis altitude of 500 km at Mars, while observing the planetary quarantine constraint and using the postulated control accuracies.

The preceding analysis was based on a single planetary vehicle approaching Mars. The effect of the presence of another planetary vehicle, a shroud cylinder, the spent S-IVB, and separation ejecta must be considered. These items affect only the selection of the initial aiming point. There is a finite probability that both spacecraft will fail to make midcourse maneuvers and hence will both arrive at the RT plane simultaneously. This probability is small, however, and has a negligible effect on aiming point selection. Therefore, it will be assumed hereafter that only one planetary vehicle will approach Mars at a given time. The shroud cylinder and the spent S-IVB, however, will accompany the reference planetary vehicle to Mars encounter. Calculations have been made of the trans-Mars trajectories of the planetary vehicle, the S-IVB stage, and the shroud cylinder. Differences in accelerations due to solar radiation pressure have been included in the calculations. The mean positions of the two extraneous bodies (i.e., S-IVB stage and shroud) are very close to the major axis of the dispersion ellipses of the planetary vehicle (a few thousands of kilometers). Dispersions of the planetary vehicle about its mean position will be accompanied by corresponding and almost identical dispersions of the other vehicles about their mean locations. Consequently, it is possible to select aiming points on the RT plane in such a way that the presence of the spent S-IVB stage and the cylinder will have only an insignificant effect on the probability of contaminating the planet. The presence of the separation ejecta complicates the problem. However, a relatively large ΔV is applied in the first midcourse correction in order to achieve aiming point biasing. Consequently, a relocation of the initial aiming point in order to accommodate the ejecta will result in only a small additional velocity increment. The increase in the subsequent errors due to control inaccuracies will therefore be small.

The previous analysis ignored the lower bound on the ΔV impulse magnitude which can be used at the second and third midcourse maneuvers. A realistic minimum impulse bit could be readily accommodated by biasing the aiming point of the third midcourse sufficiently to require a third midcourse maneuver in excess of 1 m/sec.

In conclusion, three midcourse maneuvers are adequate to permit final targeting directly to the baseline Mars orbit without violating the contamination constraint. The maneuvers are timed approximately at 5 days and 25 days from injection, and 40 days prior to encounter (this timing, however, is not critical).

3.1.5 Attitude and Velocity Control Accuracies

The preceding aiming point analysis depends on the accuracy with which the planetary vehicle's maneuvers can be controlled. Three different levels of accuracy for the attitude and velocity control were examined to assess trajectory sensitivity and establish required spacecraft capability. One of these was then selected for use in the aiming point analysis above.

Models were developed that describe expected maneuver control errors for guidance and control systems ranging from an easily achieved system to one that requires a more sophisticated design. Present state-of-the-art technology and equipment were used for each design. The performance of each system was then evaluated in terms of resultant errors in the orbital elements of the spacecraft orbit around Mars.

The following equipment characteristics were considered:

<u>Point Error</u>

- Gyro drift rate
- Gyro misalignment
- Vehicle limit-cycle magnitude
- Sun and Canopus sensor accuracies
- Switching amplifier null offsets
- Gyro converted-rate quantization error
- Gyro rate voltage/frequency converter error
- Gyro scale factor error
- Thrust vector control error

<u>Δ V Magnitude Error</u>

- Accelerometer scale factor error
- Accelerometer null offset
- Accelerometer integration resolution
- Engine tailoff uncertainty

The effective pointing accuracy of the control systems considered ranged from 6 to 14 milliradians, 1σ, as indicated on the attached trade study summary sheet, Table 3.1-8. The ΔV magnitude uncertainty, which is proportional to ΔV, ranged from approximately 0.004 to 0.04%, 1σ. In the ΔV magnitude error, which is not proportional to ΔV and ranged from approximately 0.01 m/sec to 0.1 m/sec, the engine cutoff accuracy was not considered a variable. Instead, the 3σ lunar module descent engine shutdown accuracy was used. The variation in the ΔV magnitude error, therefore, is due to varying vehicle weight during the mission and the (assumed) different thrust levels for the propulsion maneuvers, i.e., insertion, midcourse correction, and orbit trim.

Navigational accuracies for midcourse corrections were determined using an orbit determination program. DSN range and range-rate data were considered. The principal a priori error for this phase was due to uncertainty in the Mars ephemeris and unpredictable trajectory perturbations during cruise. The position uncertainty due to these effects is estimated at 100 km, 1σ. For the orbital phase, a simulation was used to estimate the orbit determination accuracy. The arrival dates were analyzed and the results are summarized in Figures 3.1-36 through 3.1-38. These results are discussed below.

Table 3.1-8: ATTITUDE & VELOCITY CONTROL REQUIREMENTS

TRADE STUDY SUMMARY SHEET

SOURCE OF REQUIREMENT

TRADE STUDY NUMBER & TITLE: GUIDANCE REQUIREMENTS UPON ACCURACY OF ATTITUDE & VELOCITY CONTROL

FUNCTIONAL AND TECHNICAL DESIGN REQUIREMENTS

The Voyager planetary vehicle must accomplish the following maneuvers: (1) midcourse, (2) insertion, and (3) orbit trim.

These maneuvers must be accomplished with sufficient precision that the planetary vehicle will be guided into an orbit that will permit achieving the Voyager's scientific goals without violating planetary quarantine.

The purpose of this study is to establish the least stringent requirements on the attitude and velocity control system which will result in the maneuver accuracy required.

The approach is to postulate the existence of three different control systems (A, B, and C); these control systems represent three levels of accuracy which are all regarded as being achievable within the current state of the art. Systems C, B, and A are regarded respectively as being easily attainable, not too dificult to attain, and attainable with difficulty. The next step is to evaluate the guidance accuracy using each system in turn. The most easily attainable system that still meets the guidance accuracy requirements of the mission is then selected.

MATRIX OF DESIGN APPROACH

Attitude & Velocity Control Errors		Alternate System A	Alternate System B	Alternate System C
Attainability within state of the art		Difficult	Not too difficult	Easy
Pointing Error, milliradians,	(1σ)	6	9	14
	(3σ)	18	27	42
$\lvert\overrightarrow{\Delta V}\rvert$ Proportional, percent, midcourse	(1σ)	.03	.17	.33
	(3σ)	.01	.50	1.00
insertion	(1σ)	.005	.02	.05
	(3σ)	.015	.07	.15
$\lvert\overrightarrow{\Delta V}\rvert$ Resolution, m /sec,*	(1σ)	.005	.005	.005
	(3σ)	.015	.015	.015
Result of analysis of guidance accuracy of Voyager mission using Systems A, B, & C		Alternate System A	Alternate System B	Alternate System C
PRO:		Satisfies Requirements for mission success	Satisfies Requirements for mission success	Satisfies Requirements for mission success
		Satisfies Requirements for planetary quarantine	Satisfies Requirements for planetary quarantine	Satisfies Requirements for planetary quarantine
		Most room for growth	More room for growth	Lowest cost
CON:		Greatest cost	Greater cost	Least room for growth

ASSUMPTIONS:

- Three midcourse maneuvers
- Mars ephemeris error = 100 km
- Attitude maneuvers are
  - $\pm$ 45 degrees roll
  - $\pm$ 180 degrees yaw or pitch

*Not including engine cutoff uncertainty, which is taken as 21 lb-sec at midcourse and orbit trim, and 260 lb-sec at orbit insertion (3σ).

SELECTION

SELECTED APPROACH

Alternate System C

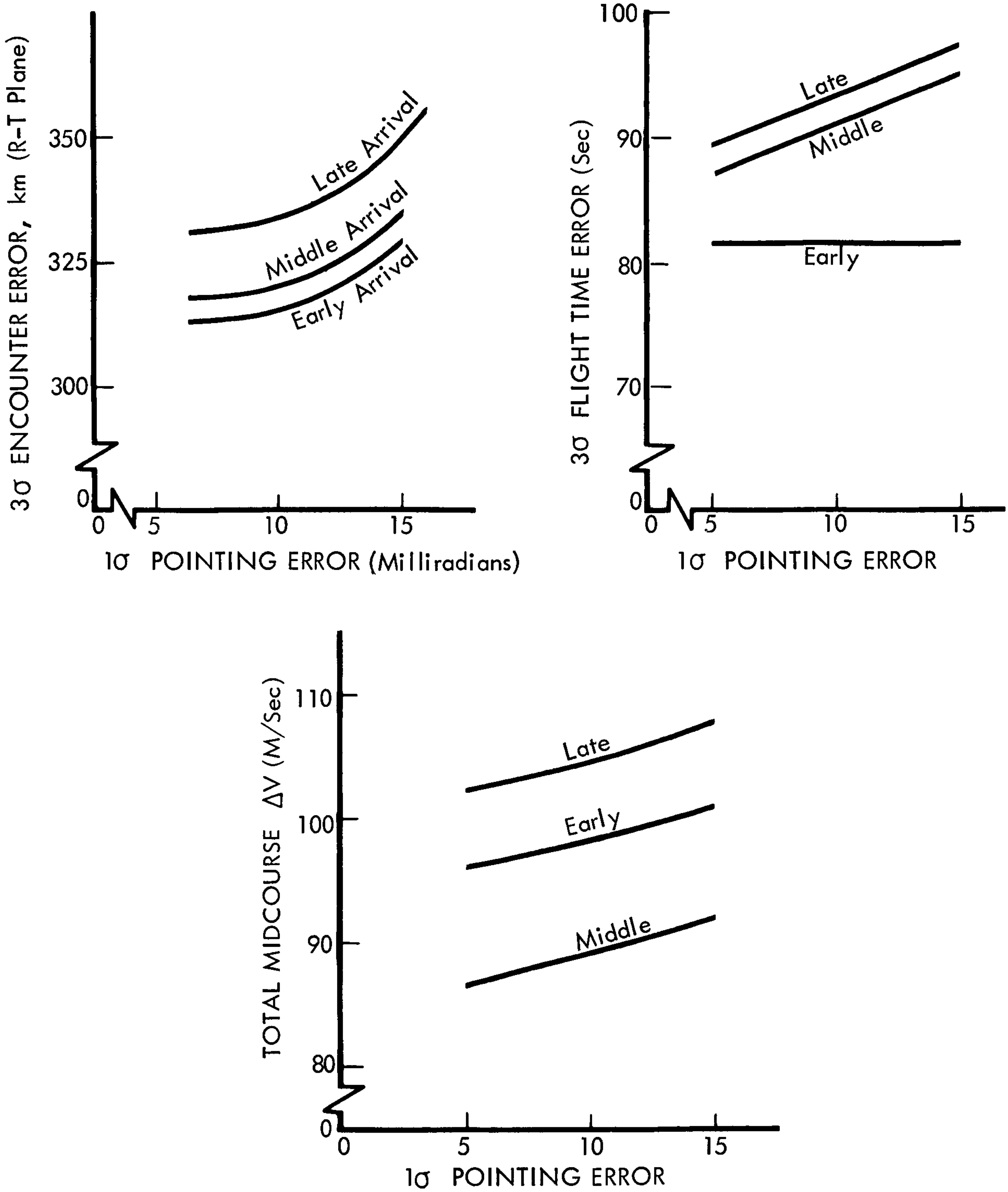


Figure 3.1- 36: MARS ENCOUNTER ERRORS AND MIDCOURSE ΔV

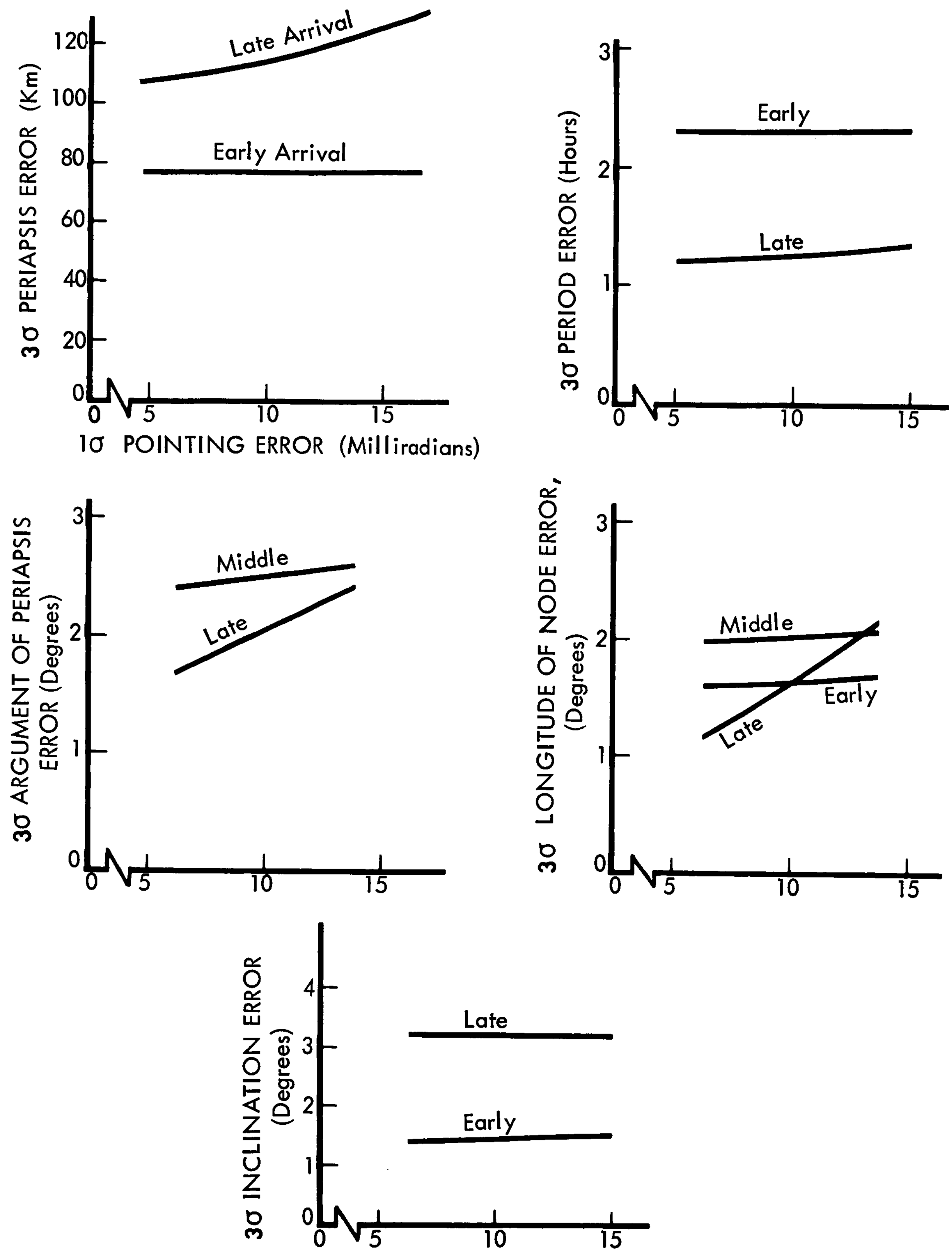


Figure 3.1-37: MARS ORBIT INSERTION ERRORS — 1973

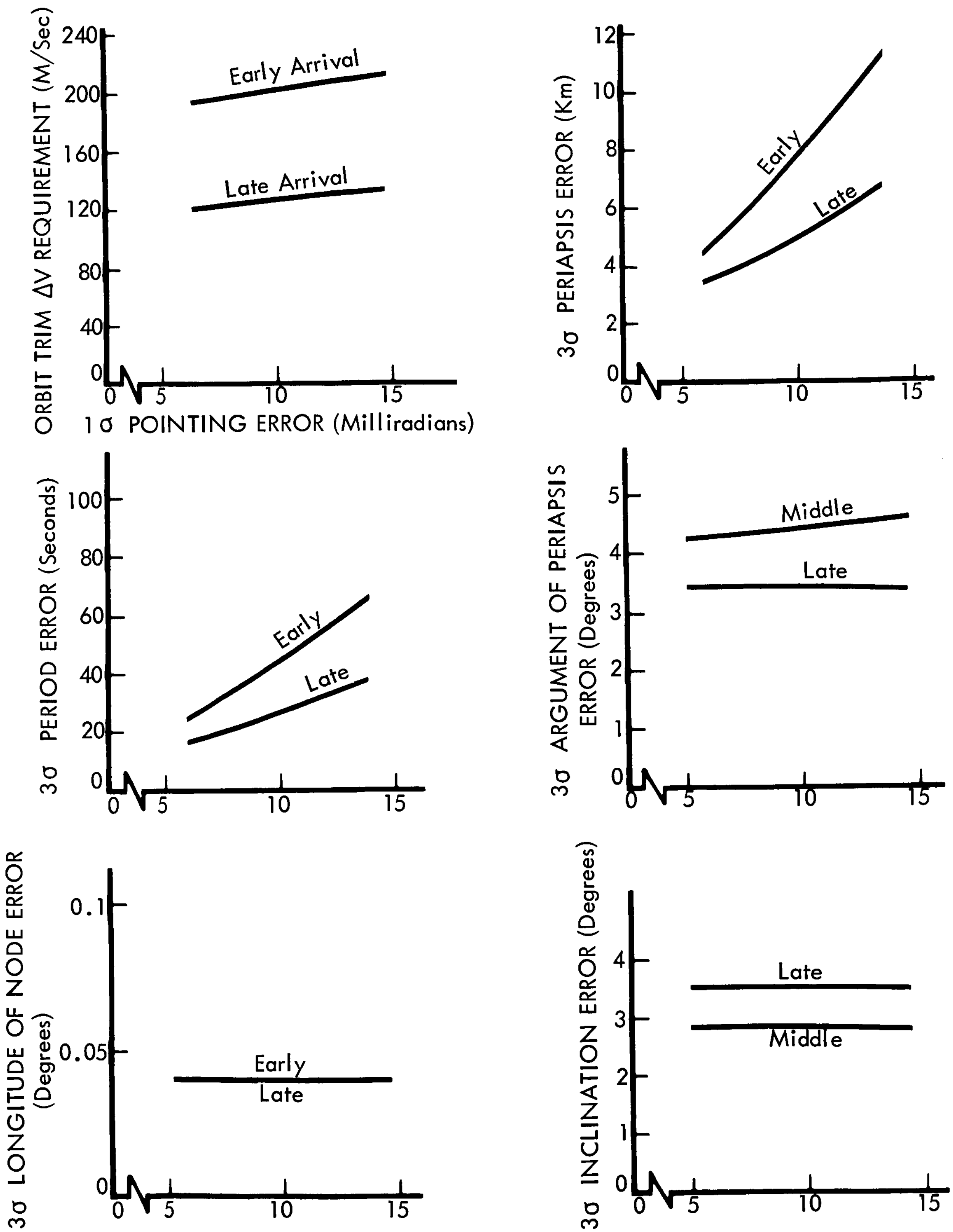


Figure 3.1-38: MARS ORBIT TRIM ERRORS AND ΔV REQUIREMENTS

Midcourse Corrections -- In the midcourse correction maneuver analysis, it was found that three maneuvers were required in order to achieve reasonable encounter accuracies and to meet the planetary quarantine constraint. The encounter errors, after three corrections and the total ΔV requirements, are shown in Figure 3.1-36 as a function of pointing accuracy and arrival date. The encounter errors are dominated by the uncertainty in Mars ephemeris and unpredictable trajectory perturbations; the ΔV requirement is dominated by the arrival time biasing requirement. In all cases, the effect of the ΔV magnitude errors, which are proportional to ΔV, were insignificant over the range of errors considered. This is true also for orbit insertion and trim. Therefore, this error source does not appear as a parameter in the plots.

Orbit Insertion -- The post-orbit-insertion errors are summarized in Figure 3.1-37 for the controlled variables (periapsis radius, period, argument of periapsis, and longitude of nodes). The dominant error source here is the Mars approach navigational uncertainty. The encounter errors, after the last midcourse maneuver, are also significant for the orbit parameters not controlled during insertion (i.e., inclination).

Orbit Trim -- The post-orbit-trim errors are given in Figure 3.1-38. For this analysis, periapsis radius, orbit period, and argument of periapsis were corrected at the first trim maneuver. At the second trim maneuver, the periapsis radius and period were held constant and the node was corrected. The total ΔV for orbit trim is also indicated in the figure.

The post-orbit-trim accuracy was evaluated in terms of imaging system resolution and photo mapping coverage and capsule delivery capability.

The posttrim errors were mapped into resolution uncertainty over a true anomaly range of 30 degrees centered about periapsis. Over this range, the variation in resolution due to the orbit trim errors was a maximum of 5%, 3σ. This level of variation in resolution will not be noticeable.

A qualitative assessment of the effects of posttrim errors on photo mapping capability was made. The results of this study were compared with the premission analysis results for Lunar Orbiter IV which performed a mapping mission at the moon. The comparison is given below.

Error	Voyager (This Study)	Lunar Orbiter IV
$\frac{1\sigma\ (\text{periapsis radius})}{\text{periapsis radius}}$	$3.7\ (10^{-4})$	$70\ (10^{-4})$
$\frac{1\sigma\ (\text{Period})}{\text{period}}$	10^{-4}	$8\ (10^{-3})$
1σ (Argument of periapsis), deg	1.4	2.0
1σ (Longitude of node), deg	0.01	0.16
1σ (Inclination), deg	1.2	0.023

The predicted Voyager accuracy compares favorably with the predicted Lunar Orbiter IV accuracies (which were verified in flight). It is expected, therefore, that the Voyager posttrim errors will have no detrimental effects on the design or operation of the mapping missions.

The most significant posttrim error from the capsule delivery standpoint is the inclination error. This result indicates that an increase in the trim ΔV is required to provide for plane change capability. Posttrim errors have an insignificant effect on the landing accuracy when compared with orbit determination and deorbit control accuracy.

Conclusions -- The following conclusions are drawn from the above analysis:

- Trajectory orbit accuracies and trim ΔV are more dependent on arrival date than on control accuracy.
- The significant orbit error sources are encounter errors due to Mars ephemeris uncertainty and unpredictable trajectory perturbations, and the Mars approach navigational uncertainty.
- Encounter errors, with three midcourse maneuvers can be held to less than 340 km, (3σ in position), and approximately 90 seconds (3σ in flight time).
- Midcourse ΔV requirements, including arrival time and aiming point biasing requirements, are less than 210 m/sec.
- Orbit trim ΔV requirement is 210 m/sec maximum.
- Posttrim errors do not present a problem for imaging resolution or photo mapping.

Based on the above results and conclusions, the Alternate System C maneuver accuracy requirements (see Table 3.1-8) are satisfactory. These are:

Error	3σ Value
Pointing	42 milliradians
ΔV Proportional	0.1%
ΔV Resolution (excluding engine)	0.02 m/sec

3.1.6 Mars Launch Opportunities in 1975, 1977, and 1979

Changes in the relative positions of Earth and Mars result in different requirements for transfer trajectories for the different launch opportunities. These requirements are indicated by a trajectory design chart for each opportunity. Figures 3.1-39 through 3.1-41 show the trajectory design envelopes for Type I trajectories in 1975, 1977, and 1979. The envelopes are based on constraints similar to those applied in 1973. The 51-degree DLA limit in 1975 and 1977 corresponds to a minimum launch azimuth of 45 degrees. In 1979, the maximum

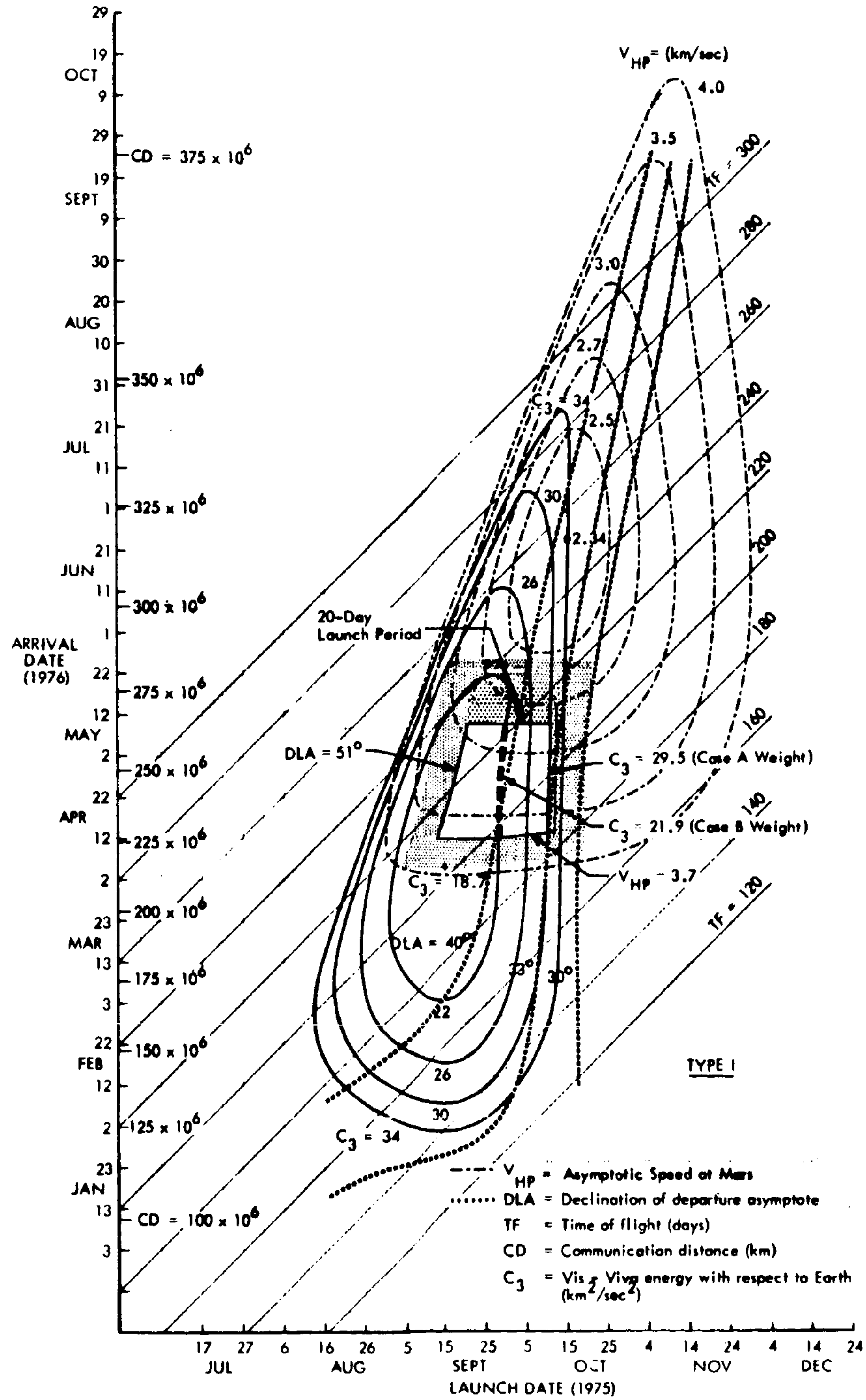


Figure 3.1-39: EARTH-MARS TRANSFER CHARACTERISTICS — 1975 TYPE I

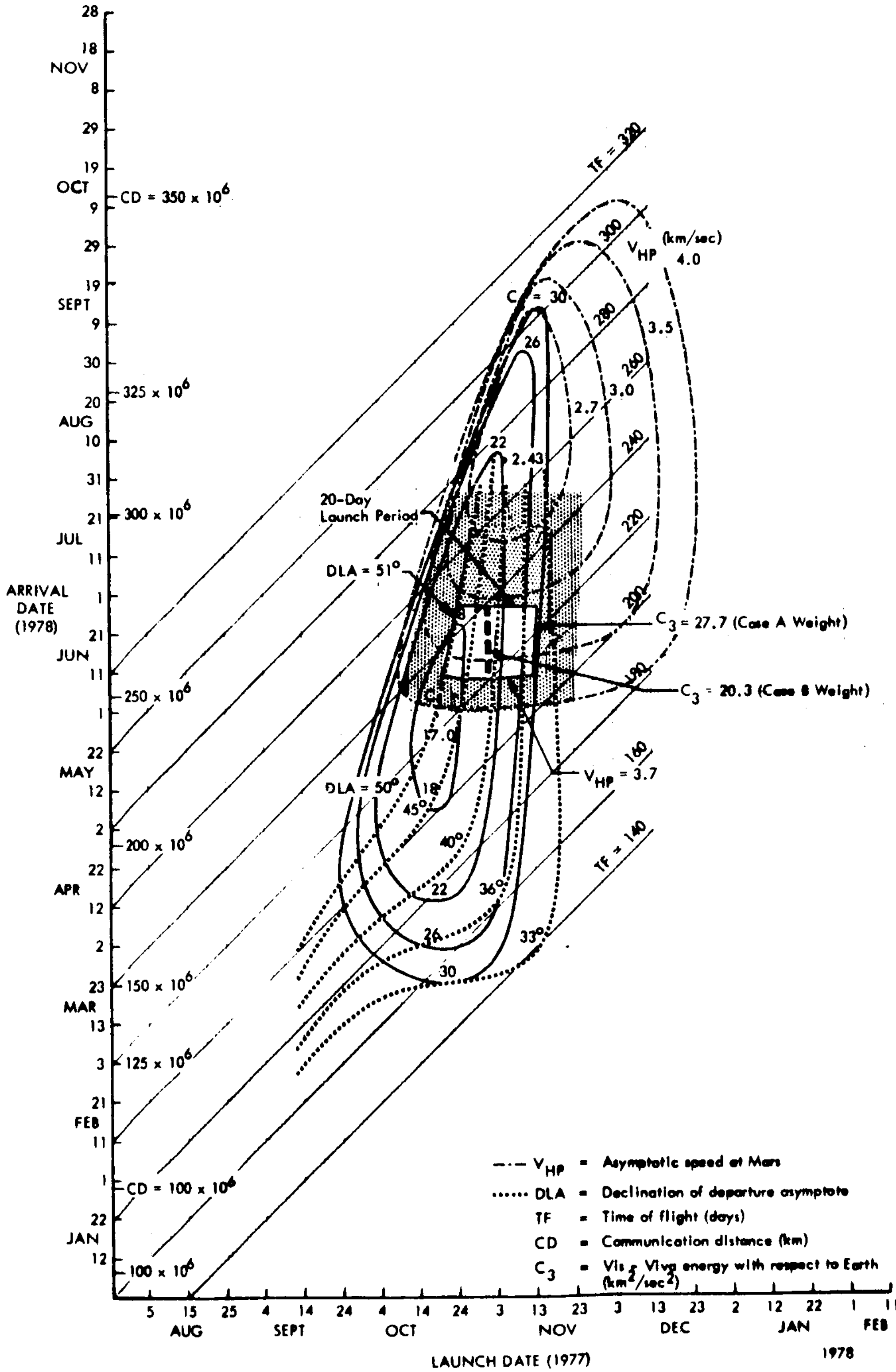


Figure 3.1-40: EARTH-MARS TRANSFER CHARACTERISTICS — 1977 TYPE I

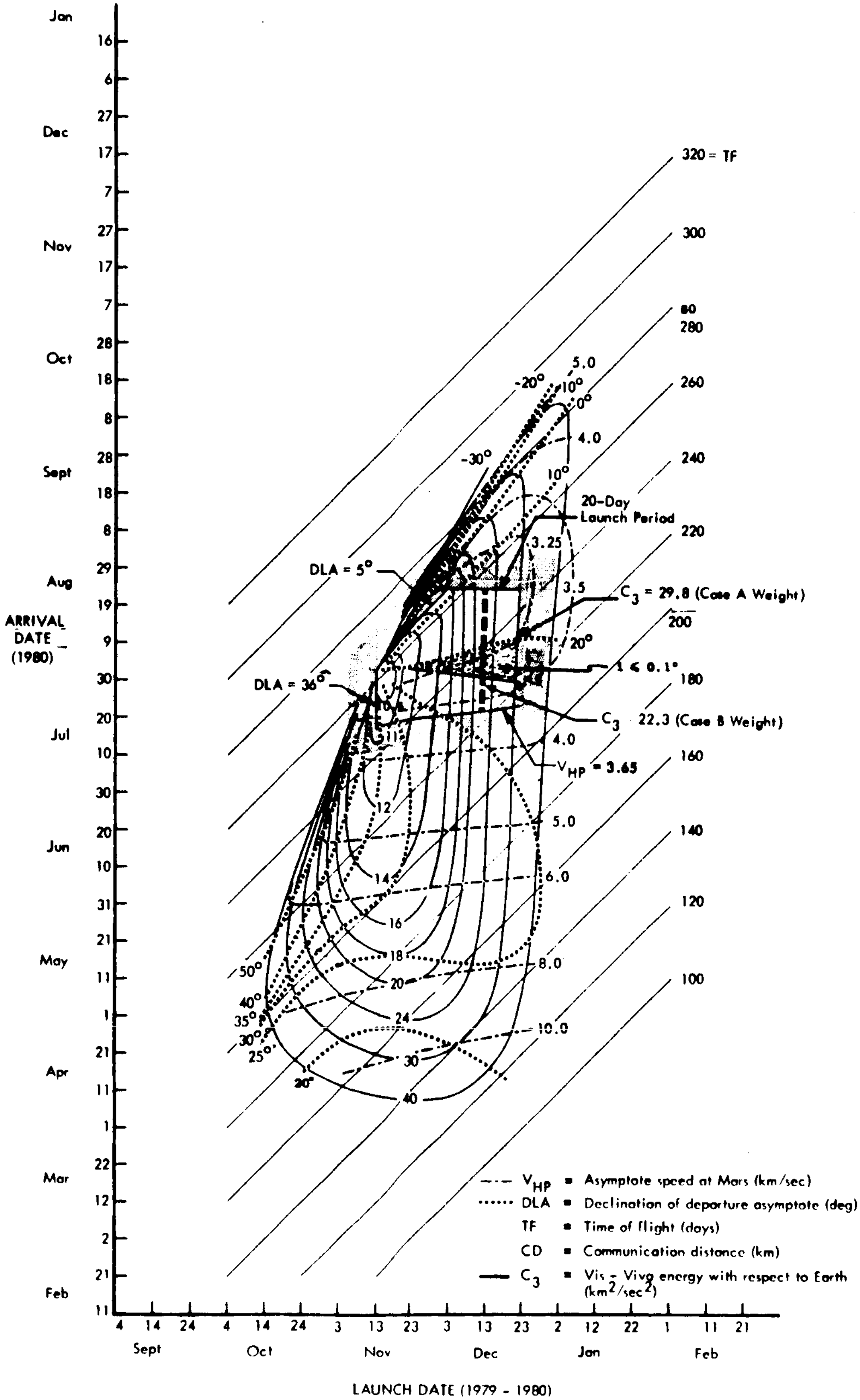


Figure 3.1-41: EARTH-MARS TRANSFER CHARACTERISTICS — 1979 TYPE I

allowable DLA has been set at 36 degrees, which corresponds to a launch azimuth of 115 degrees. Higher approach velocities, V_{HP}, are possible in the later launch opportunities because of a reduced midcourse ΔV allocation for arrival time biasing. This leaves more of the total ΔV budget of 1.95 km/sec available for orbit insertion. The target orbit, for the purpose of computing insertion ΔV requirements, and, hence, an allowable approach V_{HP}, is the 1973 baseline orbit (500 by 18,500 km, 12.4-hour orbit with a ±20 degree apsidal rotation). The target orbit is selected for the purpose of defining the range of available trajectories in these opportunities.

Comparison of the 1975, 1977, and 1979 opportunities to 1973 reveal that considerably smaller launch and arrival date regions exist for Type I trajectories. The maximum available launch periods are reduced to about half of those available in 1973. The choice of arrival dates has been similarly reduced. Communication distances at arrival, and interplanetary trip times, are similar to those available in 1973.

Type I missions in the 1975 and 1977 opportunities are difficult to perform. The C_3 and V_{HP} requirements within the design envelopes are much higher than in 1973. For a given desired launch period, the maximum payload capability of the booster will be reduced because of higher C_3 requirements and more northerly launch azimuths. Tables 3.1-9 and 3.1-10 summarize weight allocations for two specified cases. Case A, with a 5000-pound capsule, provides more C_3 capability and is indicated by the shaded limit in Figures 3.1-39, -40, and -41. Case B, which has a 7000-pound capsule in the 1975 through 1979 time period, imposes a more severe constraint on the available launch periods, as indicated by the heavy dashed line. Launch period relief is obtained by selecting Type II trajectories during the 1975 to 1977 launch periods. The Type II trajectory design charts are shown in Figures 3.1-42 and -43. Type II trajectories in 1975 and 1977 may be performed within the 90- to 115-degree launch azimuth range, allowing greater C_3 capabilities.

3.2 LAUNCH DELAY STUDY

3.2.1 Purpose

The purpose of this study was to define alternate missions that might be implemented if the scheduled 1973 launch slides beyond the full-capability launch window.

3.2.2 Scope

This study established a series of alternate Mars interplanetary trajectories and Mars orbital missions that provide launch window extensions. To provide this launch window extension, the payloads and mission profiles of the alternate missions may result in reduced system capability for meeting the mission objectives. However, such alternate conditions would allow performance of a mission several weeks after the full-capability launch window, instead of necessitating a 25-month wait for the next Mars launch opportunity.

Table 3.1-9: TARGET WEIGHT ALLOCATIONS (Pounds)

	MISSION YEAR			
	1973	1975	1977	1979
Total Planetary Vehicle	22,750	25,350	26,250	26,250
Total Dry Flight Spacecraft	5,800	6,150	6,600	6,600
Propulsion system inerts	(2,100)	(2,100)	(2,100)	(2,100)
Spacecraft bus and equipment*	(3,030)	(3,110)	(3,200)	(3,200)
Orbital science**	(390)	(650)	(1,000)****	(1,000)****
Contingency (5% dry weight)	(280)	(290)	(300)	(300)
Usable Propellants	10,950	12,200	12,650	12,650
Flight Capsule***	6,000	7,000	7,000	7,000
Planetary Vehicle Adapter Allowance	250	250	250	250
Boosted weight per planetary vehicle	23,000 lb	25,600 lb	26,500 lb	26,500 lb

* Includes 50-lb capsule bus support equipment plus 120-lb planet scan platform and science booms in 1973 and 1975.
** Includes instruments and DAE only.
*** Includes sterilization canister of which 650 lb may remain with the spacecraft.
**** 650-lb fixed mount camera system replaces scan platform.

Table 3.1-10: TARGET WEIGHT ALLOCATIONS — Alternate Flight Capsule Weights

	MISSION YEAR			
	1973	1975	1977	1979
Total Planetary Vehicle	20,750	21,450	22,250	22,250
Total Dry Flight Spacecraft	5,650	6,000	6,450	6,450
Propulsion system inerts	(2,050)	(2,050)	(2,050)	(2,050)
Spacecraft bus and equipment*	(2,940)	(3,020)	(3,110)	(3,110)
Orbital science**	(390)	(650)	(1,000)****	(1,000)****
Contingency (5% dry weight)	(270)	(280)	(290)	(290)
Usable Propellants	10,100	10,450	10,800	10,800
Flight Capsule***	5,000	5,000	5,000	5,000
Planetary Vehicle Adapter Allowance	250	250	250	250
Boosted weight per planetary vehicle	21,000 lb	21,600 lb	22,500 lb	22,500 lb

*Includes 50-lb capsule bus support equipment plus 120 lb planet scan platform and science booms in 1973 and 1975.
**Includes instruments and DAE only.
***Includes sterilization canister of which 650 lb may remain with the spacecraft.
****650-lb fixed mount camera system replaces scan platform.

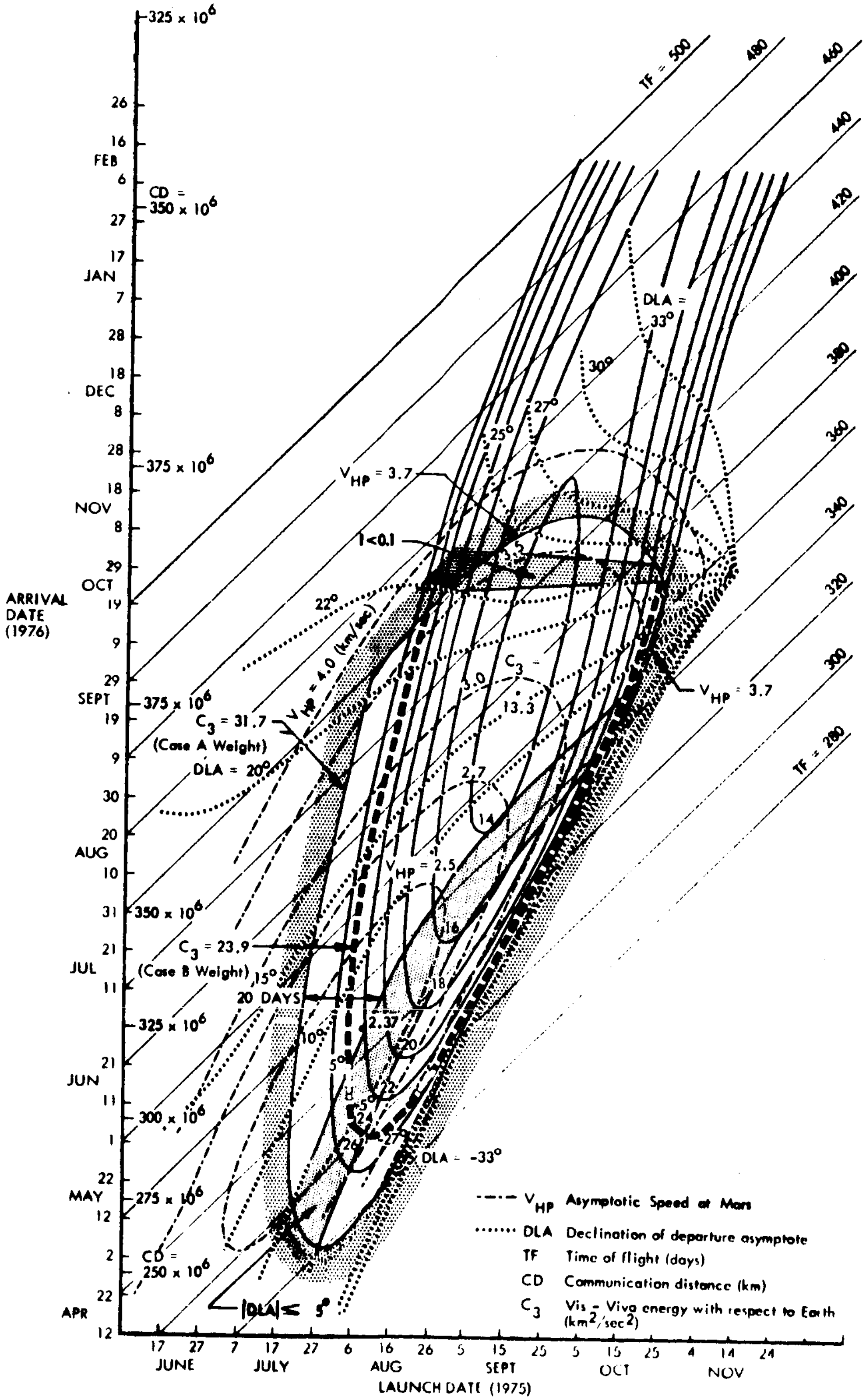


Figure 3.1-42: EARTH-MARS TRANSFER CHARACTERISTICS — 1975 Type II

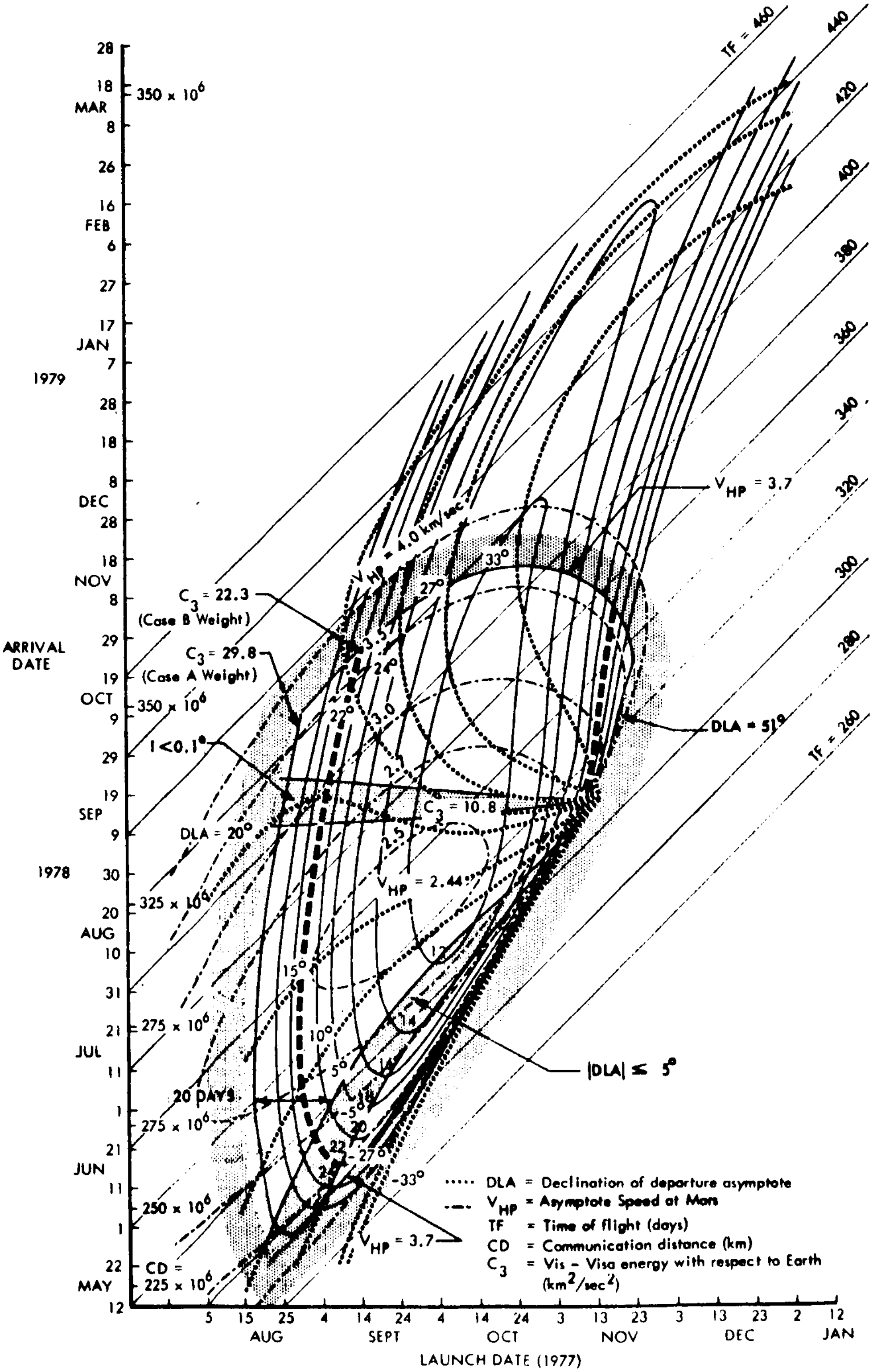


Figure 3.1-43: EARTH-MARS TRANSFER CHARACTERISTICS — 1977 TYPE II

3.2.3 Analysis and Results

The mission options considered in this study include combinations of the following:

- One or two spacecraft
- One or no capsule
- Nominal or marginally stable orbits.

For this and the following studies, representative mission and system parameters were used in lieu of the baseline mission/system parameters. The orbits considered throughout this study have a periapsis altitude of 1000 km. The nominal orbit, with a period of 13.8 hours, provides an apsidal rotation of ±20 degrees and a trim ΔV capability of 150 m/sec. The 60-hour (marginal) orbits have no provision for apsidal rotation and orbit trim. The propulsion system assumed for this study has a total ΔV capability of 2.12 km/sec.

The following system weights in pounds were used in this study.

Planetary Vehicle		21,700
Flight capsule	5,000	
Spacecraft bus and science	2,900	
Propulsion module	13,800	
Inerts	2,700	
1973 propellants	11,100	
Full propellant capacity	13,100	
Planetary Vehicle Adapter		1,500
Shroud		7,900
Jettisoned in parking orbit	4,500	
Carried to trans-Mars injection	3,400	

An envelope of allowable launch and Mars arrival date combinations for 1973 Type I trajectories is shown in Figure 3.2-1. An extension of the data to higher levels of C_3 and V_{HP} is included for use in the selection of launch dates beyond the nominal dates.

A summary of the alternate mission conditions which are most effective in extending the launch period is given in Table 3.2-1. Included in the table, for reference, is the case representing the latest launch date within the nominal trajectory envelope. The planetary vehicle weight listed on the table represents the lightest

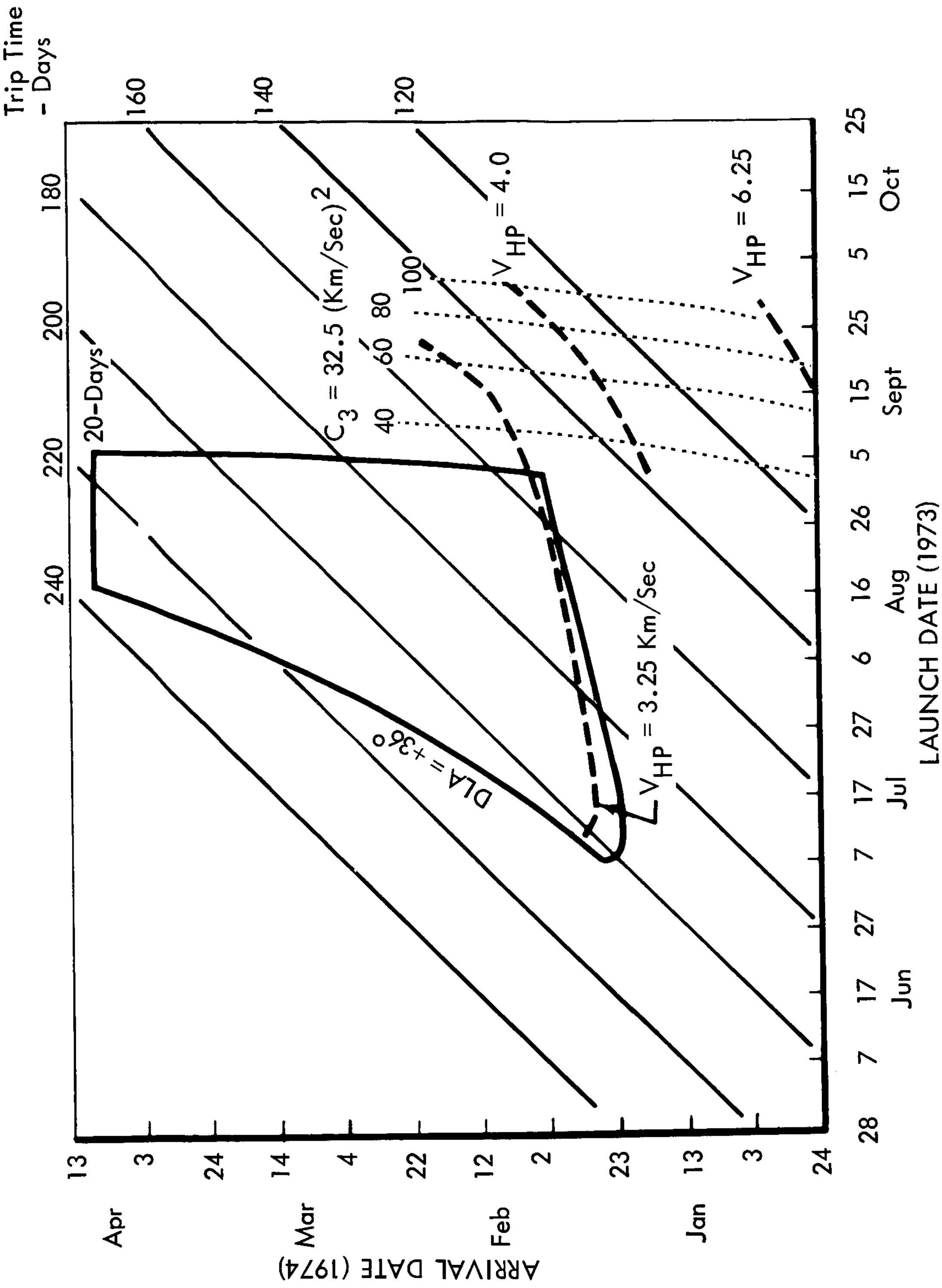


Figure 3.2-1: TYPE I EARTH-MARS TRAJECTORY PARAMETERS

TABLE 3.2-1 SUMMARY OF ALTERNATE MISSION MODES

	REFERENCE MISSION	ALTERNATE MISSION MODES					
NUMBER OF SPACECRAFT	2	2	2	1	2	1	1
NUMBER OF CAPSULES **	2	2	2	1	0	0	0
ORBIT PERIOD, HOURS	3.1	13.8	13.8	13.8	60.0	60.0	60.0
TRIP TIME, DAYS	192	151	252	159	160	167	370
TOTAL ΔV, KM/SEC	2.120	1.875	1.732	1.875	1.410	1.410	2.979
PLANETARY VEHICLE WEIGHT, LBS. ***	21,700	20,012	19,086	20,012	9,018	9,018	15,273
V_{HP}, KM/SEC	2.54	3.25	3.00	3.25	3.25	3.25	5.5
C_3*, KM^2/SEC^2	32.5	43.5	46.0	74.5	75.5	97.5	83.5
LATEST LAUNCH DATE	9/6/73	9/11/73	9/22/73	9/28/73	9/29/73	10/1/73	11/12/73
TOTAL LAUNCH PERIOD,	58	63	74	80	81	83	125
LAUNCH PERIOD INCREASE, DAYS	0	5	16	22	23	25	67

* SA-506 LAUNCH VEHICLE
LAUNCH AZIMUTH 110°
● PERIAPSIS ALTITUDE 1,000 KM

** 5000 LB CAPSULE

*** DOES NOT INCLUDE 5000-LB PROJECT PAYLOAD CONTINGENCY allowable weight compatible with the desired orbit and V_{HP}. Propellants not required for orbit insertion are offloaded prior to launch to allow for the largest possible C_3.

For a minimum trip time, the largest launch period increase is 25 days. This is the increase associated with placing a single spacecraft without capsule into a 60-hour Mars orbit period. A more desirable alternative is the 22-day extension obtained with a single planetary vehicle placed in the 13.8-hour orbit. Additional launch period extension is possible with longer trip times by using a Type II trajectory. Type II trajectory envelope data, however, are not included in this study.

3.2.4 Conclusions

Significant conclusions from this study are:

1) For normal Type I transfer trip times, the launch period can be extended 25 days by placing a single spacecraft without the capsule into a large (marginal) orbit.

2) A 22-day launch period extension is possible with a planetary vehicle (spacecraft and capsule) placed in a 13.8-hour orbit.

3) Additional gains of up to a total of 67 days are attainable by launching one spacecraft, inserting it into a marginal orbit, and increasing trip time by approximately 200 days.

3.3 MISSION ENERGY BALANCE STUDY

3.3.1 Purpose

The purpose of this study was to determine the maximum attainable weight in Mars orbit using the current launch vehicle payload capability.

3.3.2 Scope

The study was to determine the weight that could be placed in Mars orbit by balancing the performance allocation between the launch vehicle capability and the spacecraft propulsion capability. The 1973, 1975, 1977, and 1979 Mars launch opportunities were examined using a SA-506 Saturn V launch vehicle. Type I trajectories were evaluated for all launch opportunities and Type II trajectories for the 1975 and 1977 launch opportunities. The study was limited to the baseline Mars orbit (500 by 18,600 km).

3.3.3 Analysis

For each mission there is a range of trajectory parameters that, when bounded by various constraints, results in a trajectory design envelope as shown in Figure 3.3-1. The selection of any single point within the envelope results in a specific set of related trajectory parameters. These are launch injection energy (C_3), hyperbolic approach velocity at Mars (V_{HP}), launch date, and arrival date. The trajectory parameter envelope can be subdivided into 20-, 30- and 40-day launch period envelopes as shown in Figure 3.3-2. For any launch period and related

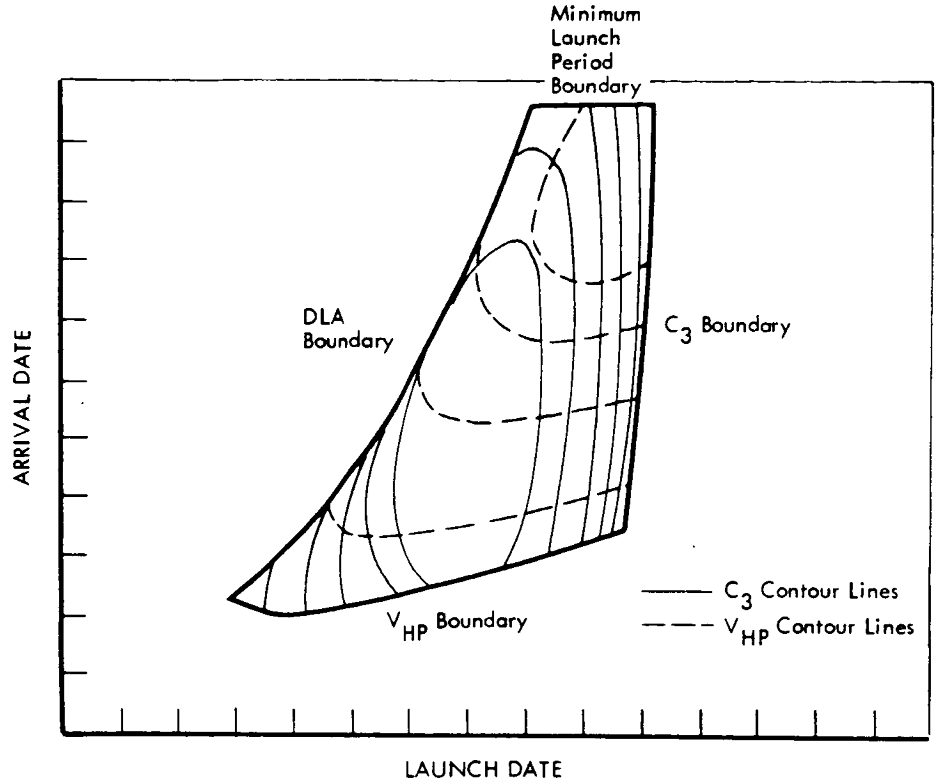


Figure 3.3-1: TYPICAL TRAJECTORY PARAMETERIC ENVELOPE

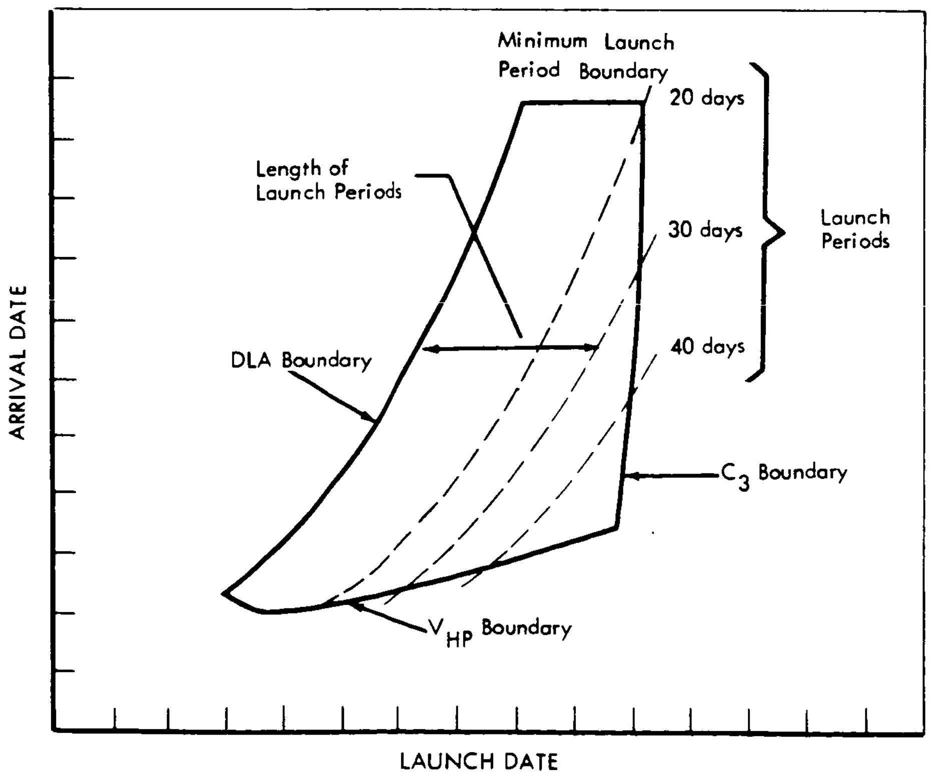


Figure 3.3-2: LAUNCH PERIOD ENVELOPES

arrival date, maximum C_3 and V_{HP} can be determined by combining the data shown in Figures 3.3-1 and 3.3-2. With these maximum values, a parametric curve can be plotted as shown in Figure 3.3-3. This curve relates maximum C_3 and V_{HP} for a given arrival date and a specified launch period. Using the data from this curve, the maximum weight in orbit was determined for each arrival date and launch period. Maximum weight-in-Mars-orbit curves for the 1973, 1975, 1977, and 1979 launch opportunities and Type I trajectories were developed as shown in Figure 3.3-4. Similar data, developed for 1975 and 1977 launch opportunities using Type II trajectories, are shown in Figure 3.3-5.

In Figure 3.3-6, the required planetary vehicle in-orbit weights for both Case A and Case B weights for the 1973-1979 launch opportunities (Section 2.0) are compared to the maximum in-orbit weight capability for the Type I trajectories. For the 1975 and 1977 launch opportunities, the required weight exceeds the capability for Case B, i.e., a 7000-pound capsule planetary vehicle configuration. This condition may be corrected by using (1) a Type II trajectory (with resultant increased mission time of 38% in 1975 and 26% in 1977); (2) an uprated Saturn V; or (3) by decreasing the project weight contingency to a maximum of 3000 and 800 p pounds for the 1975 and 1977 launch opportunities, respectively.

3.3.4 Conclusions

Significant conclusions from the study are:

1) The planetary vehicle weight in Mars orbit is sensitive to arrival date and launch period.

2) Significant in-orbit weight capability can be realized for 1973 arrival dates in mid-February. The additional capability might be used to improve reliability through further redundancy and to provide additional maneuver capability in Mars orbit (i.e., increased trim propellants).

3) For a common 1973-1979 spacecraft design using Type I trajectories, a minimum launch period of 20 days, and a 5000-pound weight contingency, weight in orbit is limited by the 1975 and 1977 missions to a maximum of 13,000 pounds. If Type II trajectories are used for 1975 and 1977, the 1979 Type I mission design limits the common spacecraft design to a maximum weight-in-orbit of 16,800 pounds.

3.4 ENCOUNTER COMMUNICATION DISTANCE STUDY

3.4.1 Purpose

The purpose of this study was to define trans-Mars trajectories and associated Mars orbits attainable with the 1973 Voyager spacecraft that minimize the Earth-to-Mars communication distance.

3.4.2 Approach

The amount of spacecraft data transmitted to Earth is limited by the amount of Earth-received RF energy. This energy varies as the inverse square of the Earth/spacecraft separation distance. Therefore, for maximum transmission capability, the mission might be planned to minimize this distance. This can be achieved by

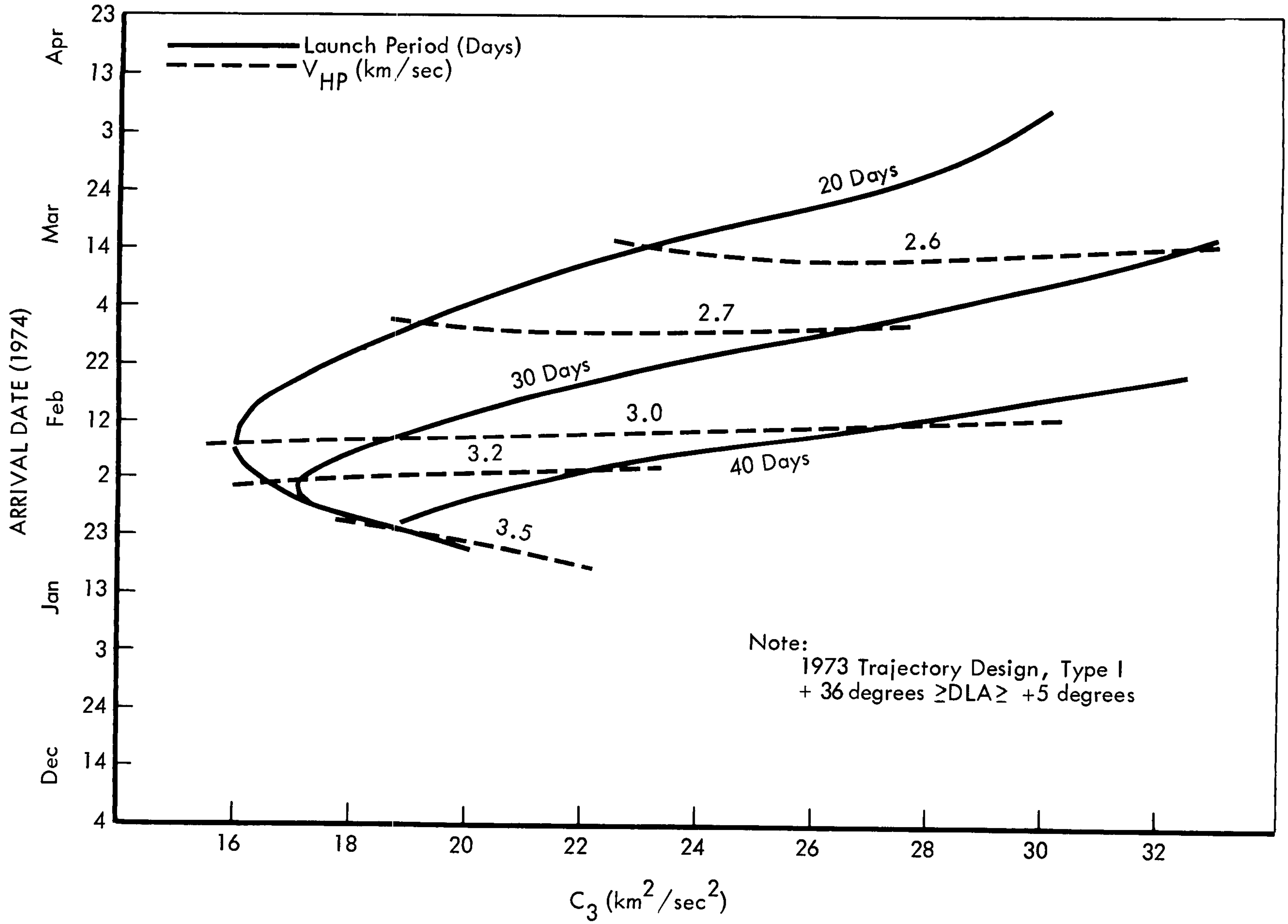


Figure 3.3-3: MAXIMUM VALUES OF C_3 AND V_{HP} — 1973 Mission

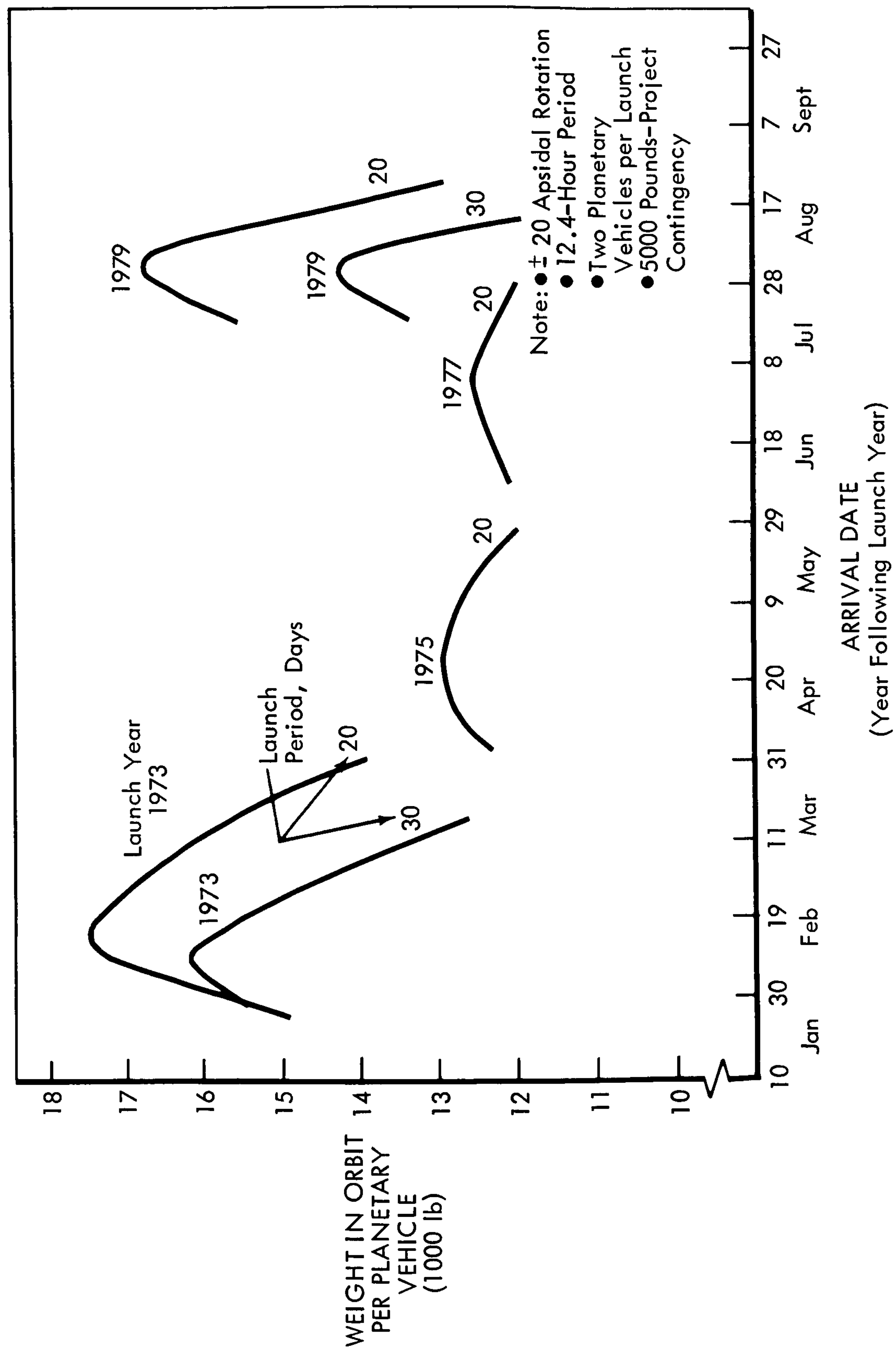


Figure 3.3-4: MAXIMUM WEIGHT CAPABILITY IN MARS ORBIT — Type I Trajectory

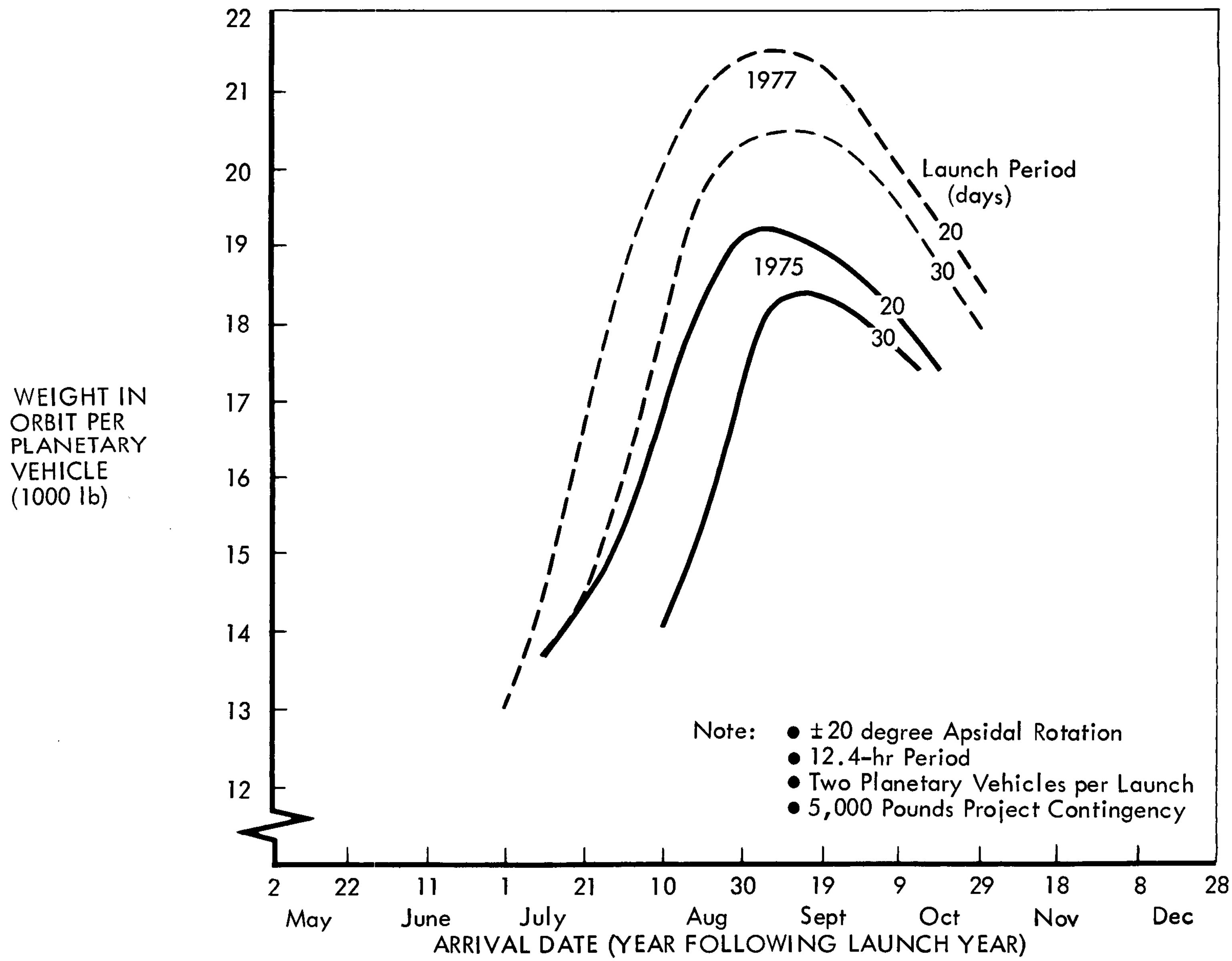


Figure 3.3-5: MAXIMUM WEIGHT CAPABILITY IN MARS ORBIT — Type II Trajectory

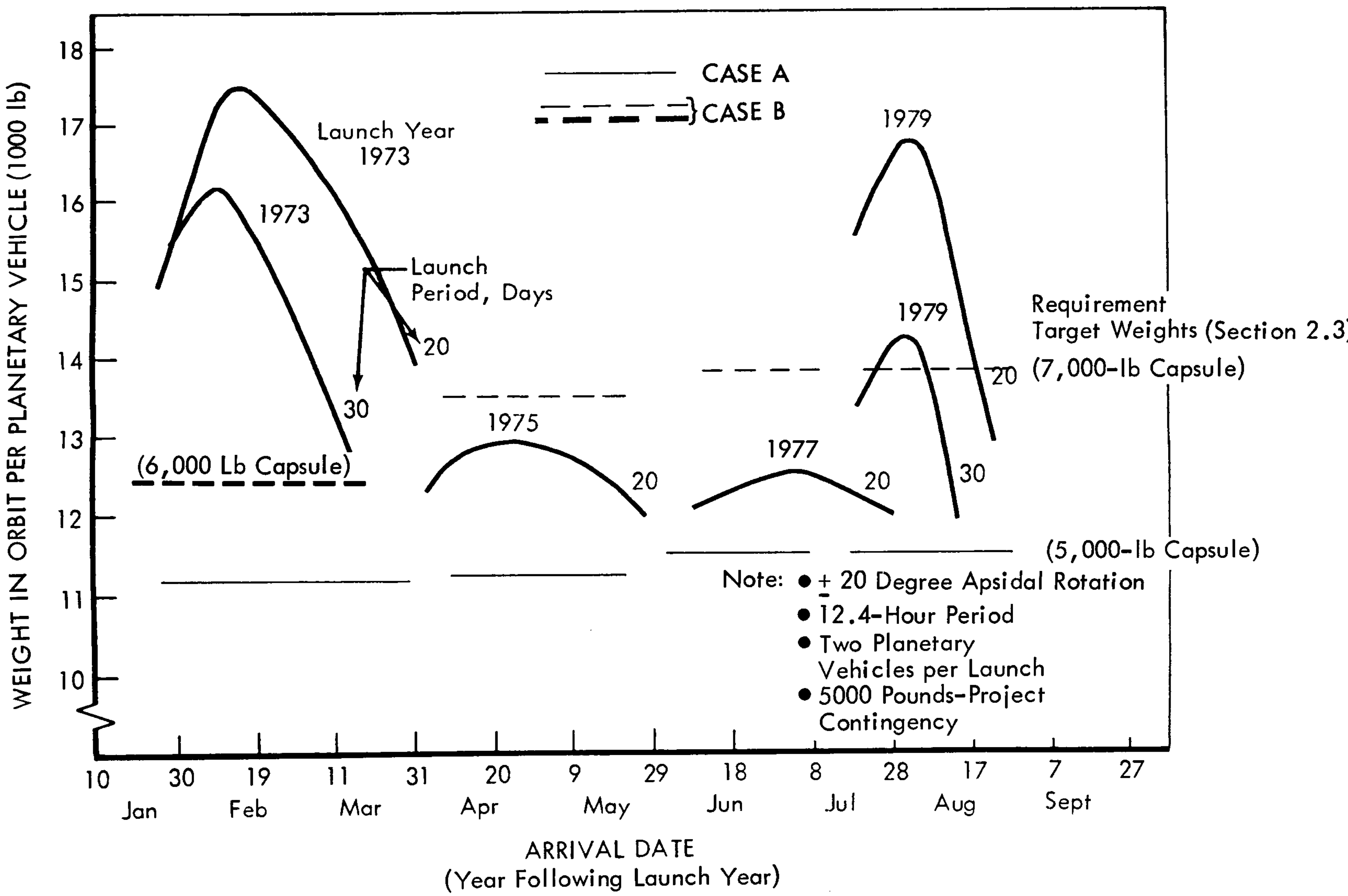


Figure 3.3-6: COMPARISON OF REQUIRED AND MAXIMUM ACHIEVABLE WEIGHTS IN MARS ORBIT — Type I Trajectories

using early launch dates, and launch vehicle payload energy to decrease Earth-Mars spacecraft transit time for a given launch date. This study investigated several methods of reducing encounter communication distance by the above means.

One method investigated for communication distance reduction is the use of fully-loaded spacecraft propellant tanks for the 1973 mission. The baseline spacecraft design includes a propulsion system sized for the 1979 mission and offloading for the prior missions (1973, 1975, and 1977). A second method investigated called for mission orbital operations involving two-step orbits. For this two-step orbital operation, the planetary vehicle is first inserted into an orbit with the desired periapsis altitude and a 60-hour period. The capsule would then be deployed and a spacecraft orbit transfer is performed to reduce the apoapsis altitude to the required level. This study established whether energy gains in orbit insertion could be realized that could then be devoted to shortening the communication distance.

3.4.3 Analysis and Results

Encounter communication distance is related to launch date and V_{HP} at Mars as shown in Figure 3.4-1. Using these data, communication distances were determined for direct orbit insertions (Mode A) and a two-step orbit operation involving an orbit change maneuver from a 60-hour orbit period to the desired final orbit (Mode B).

The weights in pounds used in developing these data were:

1973 Planetary vehicle (excluding adapter)		21,740
Spacecraft bus	2,600	
Spacecraft science	340	
Flight capsule (650 pounds remain with spacecraft)	5,000	
Propulsion module	13,800	
Inerts	2,700	
Propellants (Capacity = 13,100)	11,100	

Figure 3.4-2 shows a comparison of these two modes with propellant loading corresponding to a 1973 mission (i.e., 11,100 pounds). Figure 3.4-3 gives a similar comparison for a full propellant load condition (1979) flown in 1973.

The data show, for the 1973 propellant loading, that Mode B allows a shorter arrival communication distance for all cases considered. The reduction varies from 6 to 1% for orbit periods from 6 to 24 hours for a 500-km periapsis altitude. Using full propellant tanks in 1973 reduces the arrival communication distance from 9 to 15% below the reductions attained with the 1973 propellant loading.

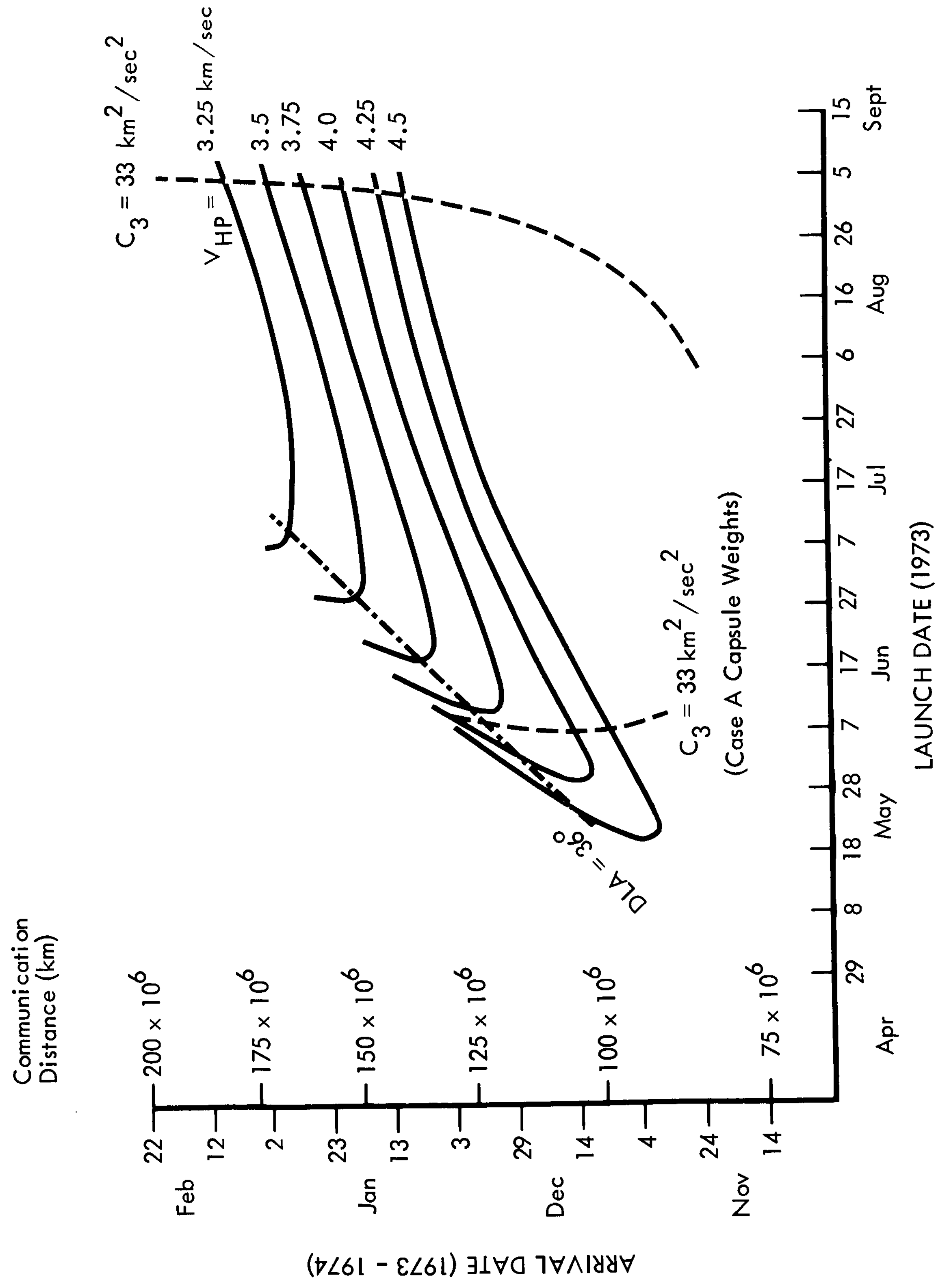


Figure 3.4-1: V_{HP} - COMMUNICATION DISTANCE RELATIONSHIP

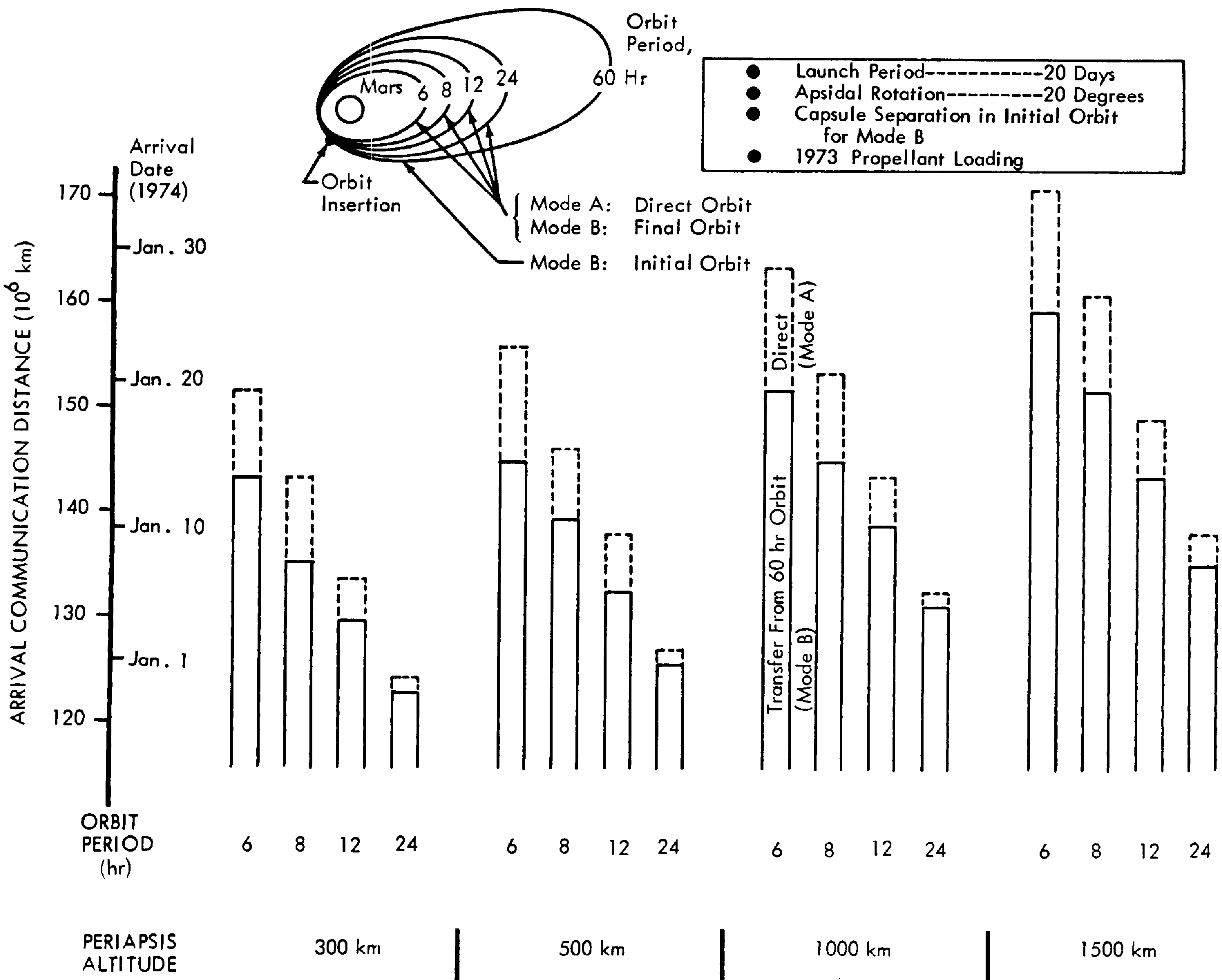


Figure 3.4-2: COMMUNICATION DISTANCES FOR 1973 S/C PROPELLANT LOADING CONDITION

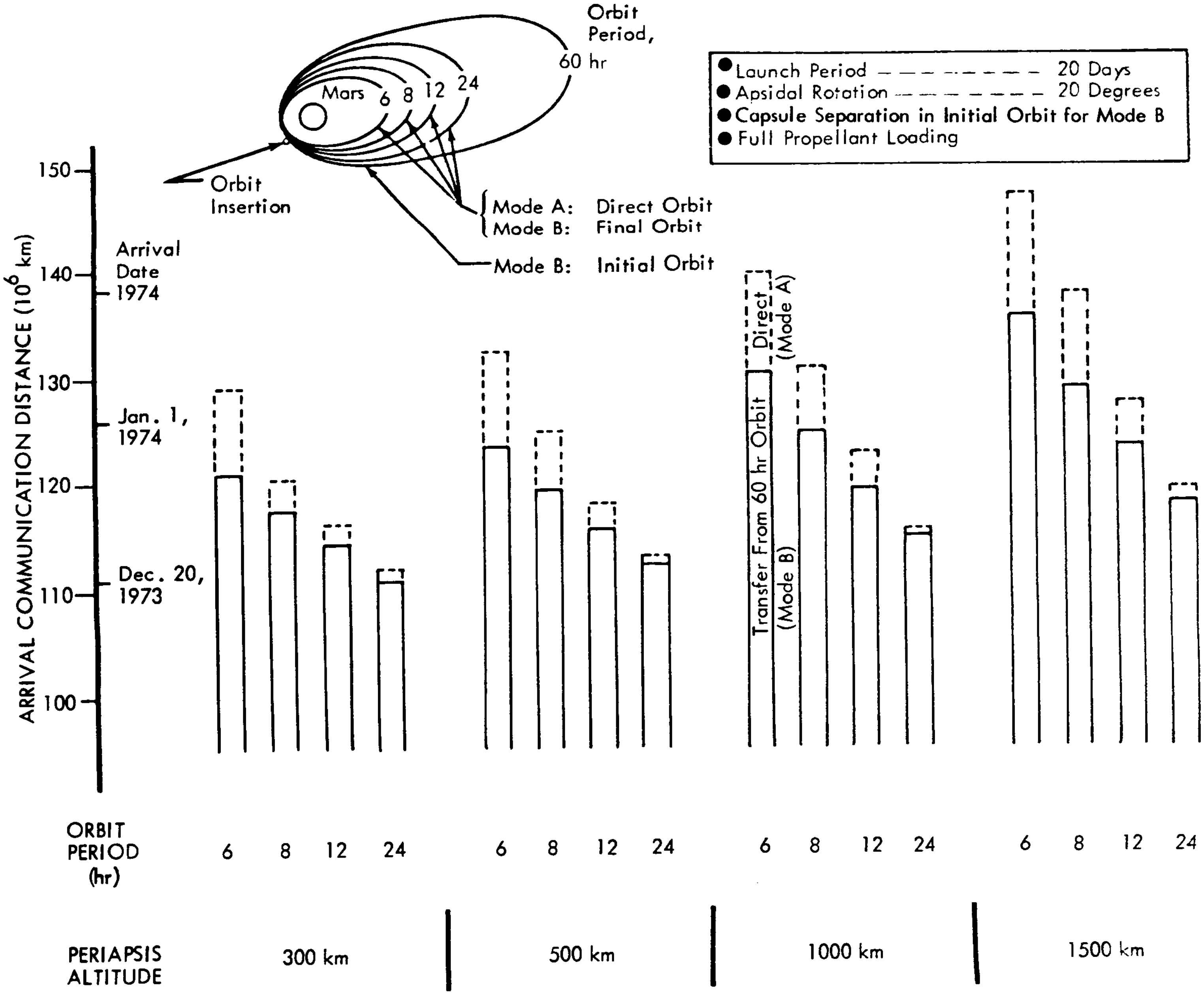


Figure 3.4-3: COMMUNICATION DISTANCES FOR A FULL S/C PROPELLANT LOADING CONDITION

In summary, the study results are as follows:

- Greatest contribution to minimizing encounter communication distance results from launching early in the opportunity. Early launch early arrival conditions (20-day launch opportunity) reduces Earth-Mars distances by about 35% over the latest April 2, 1974, arrival distance.

- The communication distance can be reduced as much as 6% by using an orbit insertion mode involving orbital transfer to the desired final orbit from an initial 60-hour orbit. The capsule must be deployed from the initial orbit to realize the energy gain for use in reducing the encounter communication distance.

- Loading the propellant tanks to capacity results in an additional communication distance reduction of as much as 15%.

3.5 CAPSULE-CANISTER SEPARATION TRADE

3.5.1 Purpose

The purpose of this study was to determine when in the mission sequence to separate the capsule forward sterilization canister from the planetary vehicle.

3.5.2 Scope

The Mars planetary vehicle consists of a flight spacecraft and a flight capsule. The capsule is sterilized and encapsulated before launch to avoid Mars contamination. The canister is unencapsulated by jettisoning before capsule separation from the spacecraft. This study was conducted to investigate where in the mission sequence this canister jettisoning should be done. Study considerations included planetary contamination constraints and orbit insertion propellant requirements.

3.5.3 Analysis and Results

Three alternate separation modes of separating the canister were examined: 10 days prior to, 1 day prior to, and X days after orbit insertion. Figure 3.5-1 presents the considerations involved for the case where the canister is separated prior to the orbit insertion. Table 3.5-1 is a summary of the trade analysis that was performed. This summary describes each canister separation mode and defines the preferred mode of canister separation.

Separation prior to orbit insertion is compatible with the planetary quarantine constraint and will result in propellant savings. However, a requirement of retaining the capsule in orbit for up to 30 days may develop. This would allow sufficient time for orbital photoimaging to aid in capsule landing site selection. Maintaining capsule thermal control for 30 days in the unencapsulated condition may prove impossible. Consequently, it is prudent to plan on jettisoning the forward capsule sterilization canister after orbit insertion and just before capsule release.

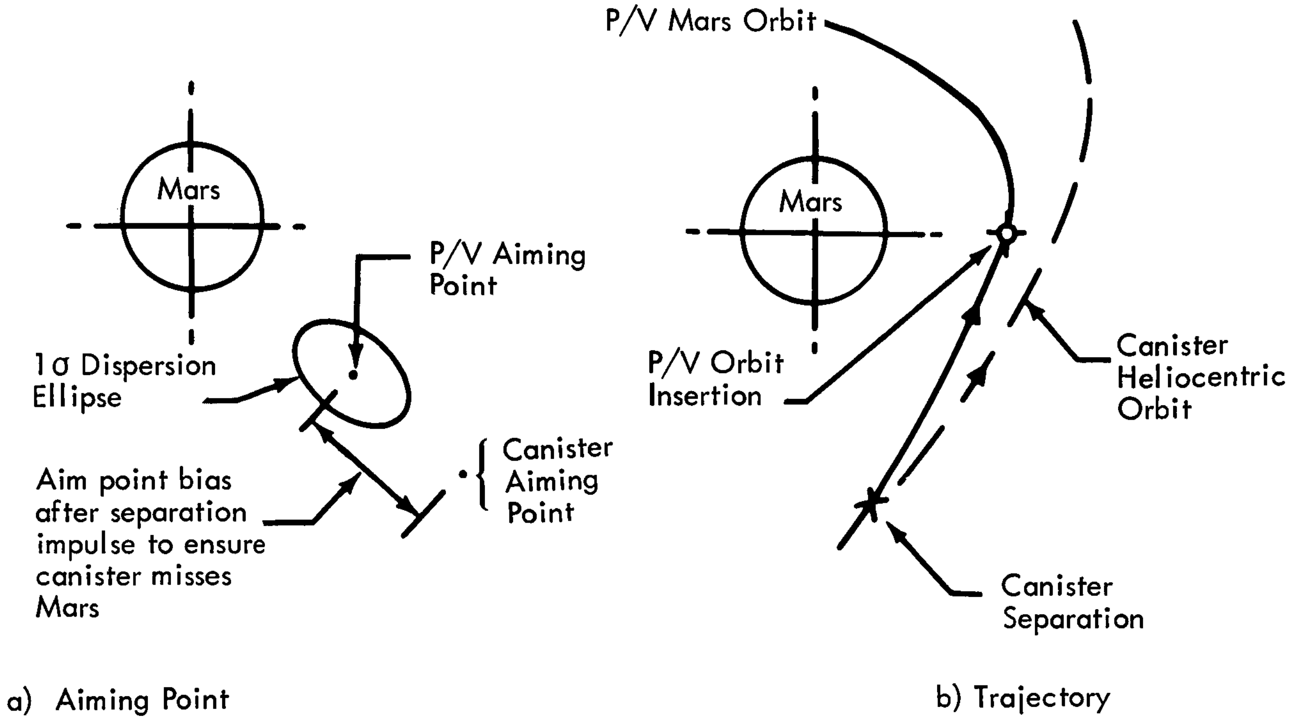


Figure 3.5-1: CANISTER/PLANETARY VEHICLE SEPARATION GEOMETRY CONSIDERATIONS

3.6 SPACECRAFT PROPULSION ENGINE SELECTION STUDY

3.6.1 Purpose

The purpose of this study was to evaluate candidate spacecraft propulsion subsystem engines for the 1973-1979 Voyager Mars missions.

3.6.2 Scope

To perform this evaluation, propulsion subsystem sizing parameters were developed for each of the 1973-1979 missions. The evaluation parameters included weight, performance, mixture ratios, reliability, and development status. Use of a common subsystem for all four missions (1973-1979), including a common propellant tankage configuration, was a study guideline. The engines evaluated were: (1) the Apollo Lunar Module Descent/Engine (LMDE), (2) the Bell Aerosystems Agena Engine (Model 8517), (3) the Titan III Transtage engine, and (4) the LMDE supplemented with Thiokol C-1 low thrust engines.

3.6.3 Analysis

Propellant weight requirements and total payload weight capability are shown in Figures 3.6-1 through 3.6-4 for the four candidate engine systems for the 1973 through 1979 Voyager missions. The candidate propulsion subsystem modules were

Table 3.5-1: CAPSULE CANSISTER SEPARATION TRADE SUMMARY

FUNCTIONAL AND TECHNICAL DESCRIPTION	SEPARATION OF CANISTER PRIOR TO INSERTION INTO ORBIT AROUND MARS		SEPARATION OF CANISTER AFTER INSERTION INTO ORBIT AROUND MARS	SELECTION
	① At least 10 days prior to insertion.	② 1 day prior to insertion	③ X days after insertion	
The Capsule canister is the biological barrier which ensures that the Capsule acquires no surface contamination prior to its descent to the surface of Mars. It is necessary to jettison a portion of the canister before the capsule can be separated from the spacecraft. Capsule diameter is assumed to be 190 inches. The purpose of this trade study is to determine when, in the mission, the canister should be jettisoned. The separation device itself is a part of the capsule system. Determine the impact on the Voyager contamination constraint as a function of when the canister is separated. Determine the impact on the flight hardware (trajectory, attitude, enssuring non-interference with S/C or sensors, etc.) as a function of when the canister is separated. Selection Criteria: 1) Probability of planetary contamination 2) Weight 3) Canister /spacecraft interface	Trajectory Considerations: Separation Impulse 0.4 m/sec Aim Point Bias 194 km Canister is placed in helio-centric orbit Contamination Probabilities Potential micrometeoroid spalling effect 10^{-14} Potential N_2 gas contamination without sterilization 0.7×10^{-5} with sterilization 0.7×10^{-7} Weight: Since canister (400 lb) is not carried into Mars orbit, a reduction of 190 lb of insertion propellants is obtained. Interface: Physically the same for all three options. Operationally exposes Capsule System to deep space cooling and micrometeroid environment until capsule separation.	Trajectory Considerations: Separation Impulse 0.5 m/sec Aim Point Bias 43 km Canister is placed in helio-centric orbit Contamination Probabilities Potential micrometeoroid spalling effect 10^{-14} Potential N_2 gas contamination without sterilization 0.7×10^{-5} with sterilization 0.7×10^{-7} Weight: Since canister (400 lb) is not carried into Mars orbit, a reduction of 190 lb of insertion propellants is obtained. Interface: Physically the same for all three options. Operationally exposes Capsule System to deep space cooling and micrometeroid environment until capsule separation.	Trajectory Considerations: Separation impulse 0.5 m/sec applied at apoapsis opposite to velocity vector. Canister/planetary vehicle separation distance at first periapsis passage - 30 km (this increases on subsequent orbits). Periapsis must be increased 90 km to ensure comparable orbit life-time (11 years) of the S/C and canister. Canister is placed in Mars orbit Contamination Consideration: No increased probability of contaminating Mars if above trajectory requirements are accommodated. Weight: The additional 190 lb of propellant will be required for insertion Interface: Physically the same for all three options. Operationally protects Capsule System from radiating into deep space and from the micrometeroid environment until canister separation.	Trajectory Considerations: Best (1) Acceptable (2) Least desirable (3) 90 km increase in periapsis alt. has a detrimental effect on the mission. Contamination Considerations: Best (3) Acceptable (1) & (2) Weight: Best (1) & (2) Least desirable (3) Interface: Best (3) Undesirable (1) & (2) SELECTED MODE: Separate canister X days after insertion

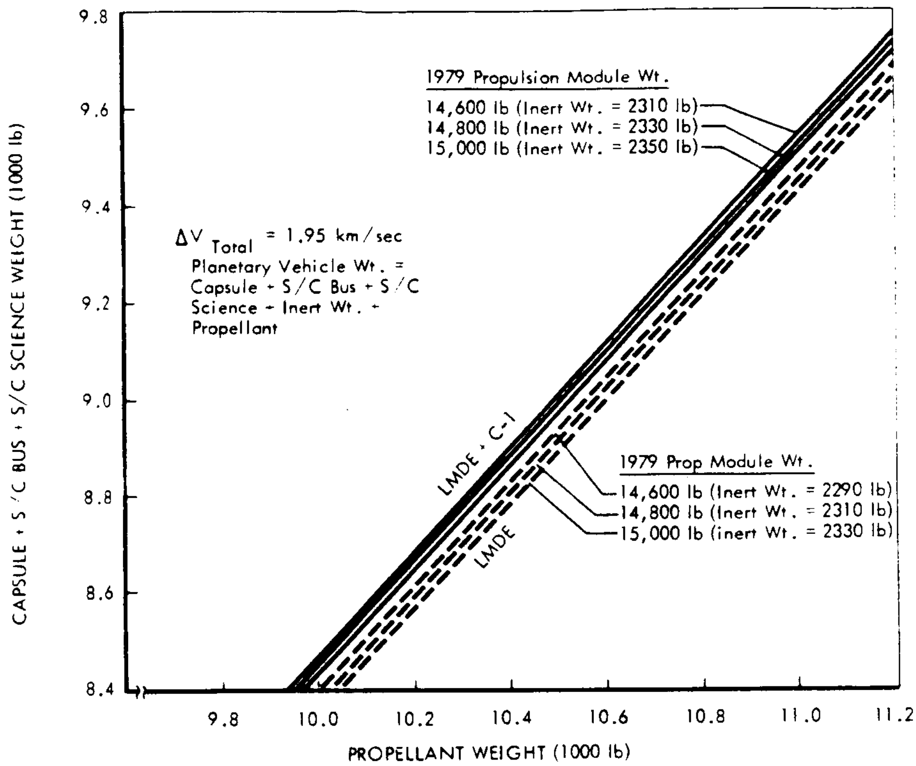


Figure 3.6-1: LMDE AND C-1 ENGINE PROPELLANT REQUIREMENTS
1973 Mission

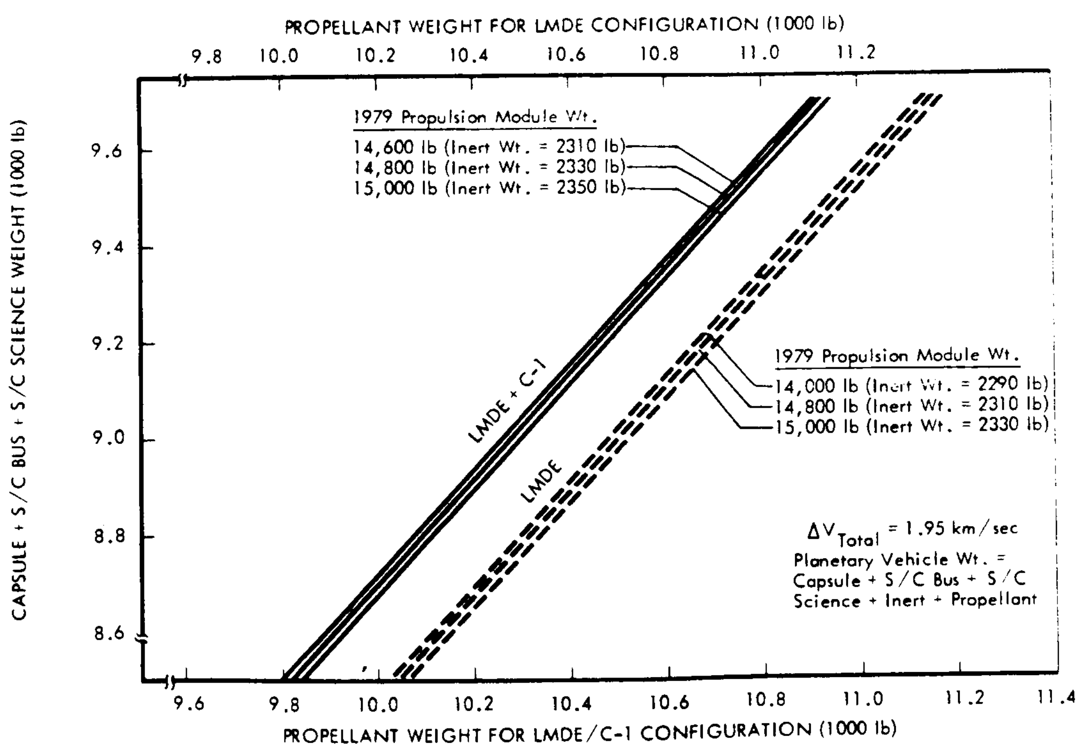


Figure 3.6-2: LMDE AND C-1 ENGINE PROPELLANT REQUIREMENTS
1975 and 1977 Missions

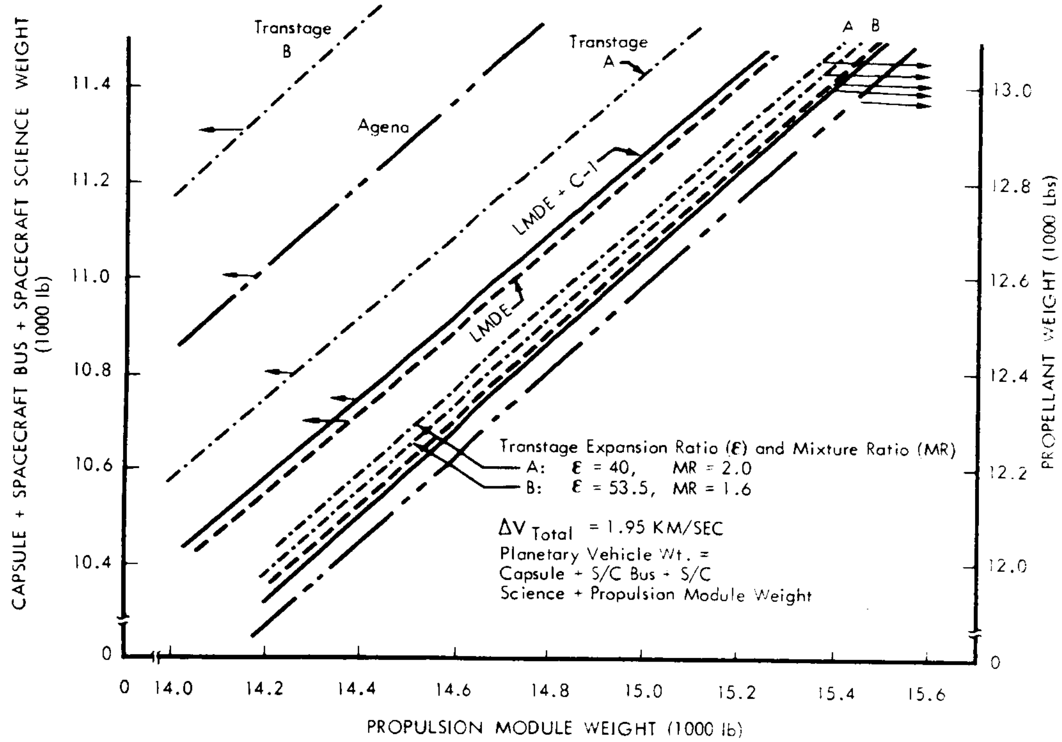


Figure 3.6-3: LMDE, LMDE/C-1, TRANSTAGE, AGENA ENGINE PROPELLANT REQUIREMENTS

1979 Mission

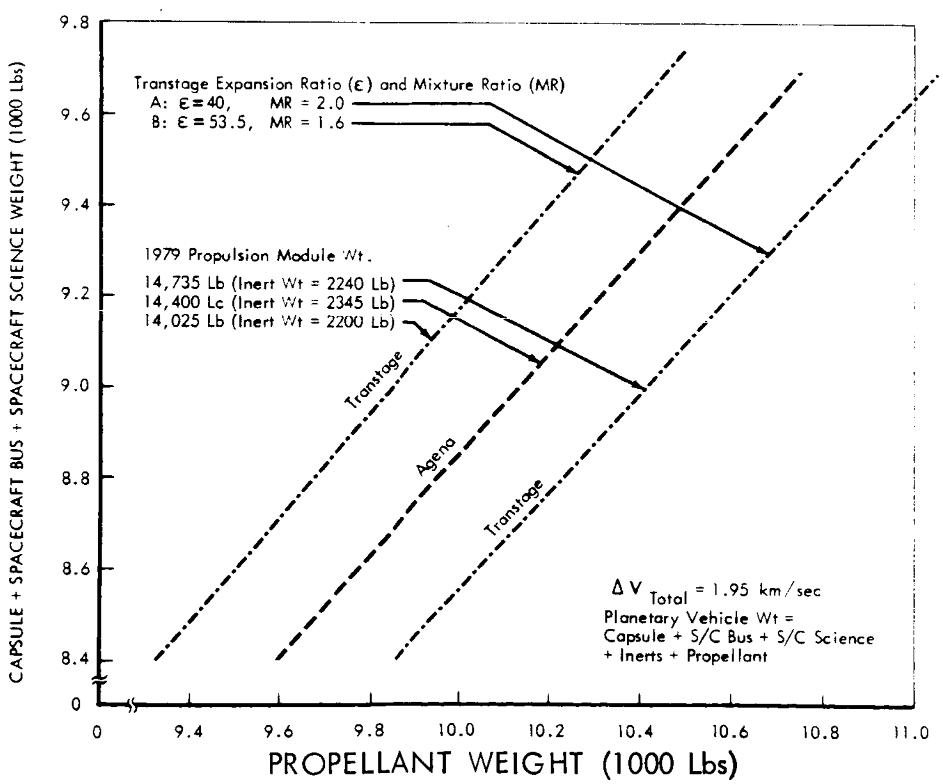


Figure 3.6-4: TRANSTAGE AND AGENA ENGINE PROPELLANT REQUIREMENTS

1973 Mission

sized to meet the most demanding requirements, which occur in 1979. Propellant loadings required for various payloads to meet the 1973-1977 mission requirements were then determined. The payloads consisted of the capsule and the flight spacecraft, including the science payload.

Propulsion module sizing was based on a constant total ΔV requirement of 1.95 km/sec for all four missions. ΔV allocations for midcourse corrections, arrival time biasing, and orbit trim were subtracted from the ΔV allocation of 1.95 km/sec to obtain the orbit insertion ΔV.

Allowance was made for variation of inert weights as a function of propellant requirements. Adjustments were also made for the differences in weights of the engines considered. However, no allowance was made for variation in inert weights as a function of engine thrust. Therefore, the weight estimates developed in this study become less accurate as engine thrust level on the alternate engine configurations deviates from that of the LMDE's high thrust.

Engine Systems -- Technical data for the engine studies were obtained from TRW Power Systems Division (LMDE), Aerojet-General (Titan III Transtage), Bell Aerosystems (Agena Model 8517), and Thiokol (RMD C-1).

The LMDE was considered for all ΔV maneuvers (in a high thrust mode of 9850 pounds for the orbit insertion maneuver and a low thrust mode of 1050 pounds for midcourse and orbit trim maneuvers). The LMDE was also considered for the orbit insertion maneuver while a cluster of four C-1 engines were used for midcourse correction and orbit trim. Because of losses due to engine throat erosion, the specific impulse, I_s, was assumed to be 304 lbf-sec/lbm during high thrust operation, and 288 lbf-sec/lbm during low thrust operation. From engine weight breakdowns supplied by TRW, the engine weight was determined to be 389 pounds.

The Titan III Transtage engine was considered to operate at a fixed thrust level ($F = 8000$ pounds, $I_s = 303$ lbf-sec/lbm) for all maneuvers. It nominally operates at a mixture ratio of 2. However, the engine shows a specific impulse increase of 6 lbf-sec/lbm when operated at a mixture ratio of 1.6. The length and maximum diameter of the Transtage engine are smaller than the LMDE. Aerojet-General has submitted data showing that, by changing the mixture ratio to 1.6 and increasing the nozzle expansion ratio to 53.5 to match the geometric envelope limits of the LMDE, the Transtage engine I_s can be increased to 315 lbf-sec/lbm. The improved engine was considered for all maneuvers. Aerojet also submitted weight estimates for an ablative nozzle skirt required by the Transtage engine to meet the maximum temperature of 1200° F specified for current Voyager design concepts. These weight estimates were 315 pounds for the engine at an expansion ratio of 40 and 360 pounds for the engine at an expansion ratio of 53.5. These weights were used in this study.

The Agena Model 8517 is an advanced engine design based on the current Agena engine concept. Studies of the use of this engine design concept in several operating modes were made for MSFC by Bell Aerosystems. The concept chosen for this study used the engine for all Voyager propulsion maneuvers. For this concept, the engine is pump-fed in a high thrust mode for orbit insertion, and pressure-fed in a low thrust mode for midcourse corrections and orbit trim. This design is also capable of operating in a pressure-fed mode at an intermediate thrust level as a high thrust mode backup. The Agena engine weight used in the study was 481 pounds. Other parameters used for the engine are:

	Thrust (lb)	Mixture Ratio	I_s (lbf-sec/lbm)
Orbit insertion	17,600	1.65	316
Midcourse and orbit trim	960	1.65	291

Data on the C-1 engine were obtained from Thiokol, RMD publications PD-5-66, Specification No. CD15166A, Revision A, and direct communications with Thiokol representatives. The C-1 engine is a family of engine designs developed for NASA/MSFC as backup engines for the Apollo program. The version used in this study has completed a qualification program for the Apollo lunar module. This engine has the following characteristics:

- Thrust = 100 pounds
- I_s = 298 nominal (either MMH or 50% N_2H_4/50% UDMH and N_2O_4)
- ε (expansion ratio) = 60
- Weight = 12 pounds with quad-redundant values.

LMDE and LMDE/C-1 Comparison -- The parametric weight data (Figures 3.6-1 through -4) show that the propulsion system weights and payload capability of the LMDE and LMDE/C-1 are nearly identical. For the 1979 Mars mission, a design using the LMDE alone is 35 pounds heavier than a propulsion module using the LMDE/C-1 engines. This results in a planetary vehicle weight savings of approximately 65 pounds for the LMDE/C-1 combination in 1973.

The manufacturer's predicted reliabilities are 0.9989 for the LMDE, and 0.9992 for the C-1.

LMDE, Transtage, and Agena Engines Comparison -- A comparison of the pertinent physical and performance characteristics of the three candidate engines is summarized in Table 3.6-1. From accuracy considerations, note that the higher the engine thrust, the greater the planetary vehicle orbit dispersions due to thrust vector errors and engine shutdown impulse tolerances. Therefore, the Agena engine is expected to result in greater errors at orbit insertion than either the LMDE or Transtage. Likewise, the Transtage engine would produce greater dispersions at midcourse corrections and orbit trim than either the LMDE or Agena engines. The Transtage engine falls short of the the 1973 Mars mission minimum impulse-bit requirement of 1 m/sec velocity change for the last midcourse correction. However, information supplied by Aerojet-General indicates that closing the propellant valves before full thrust condition is reached will yield impulse bits below the minimum Voyager requirement.

Table 3.6-1: PROPULSION SUBSYSTEM SELECTION TRADE SUMMARY

FUNCTIONAL & TECHNICAL REQUIREMENTS		SELECTION CONSIDERATIONS: LMDE	LMDE - 4(C-1) C-1 Engine Data	AGENA MODEL 8517	TRANSTAGE	SELECTION
Select the engine(s) and size the propellant loading and plumbing for the 1973-1979 Mars Voyager missions. The study shall include: 1) Determine propellant quantities required for the 1973 - 1979 Voyager Mars missions utilizing LMDE. 2) Determine the interchangeability of the Transtage and Agena engines with the LMDE. 3) Determine the suitability of the alternate LMDE + C-1 engine cluster engine configuration. Subsystem Selection Criteria: • Engine performance • Subsystem weight • Development status	**Engine Parameters:**					
	I_{sp} - sec (Chamber pressure - psia)	304(102) 288(11.3)	298(96)	316(500) 291 (28)	303(108) at MR = 2 309 at MR = 1.6	
	Thrust (lb)	9850 to 1050	100	17,600 & 960	8000	
	Propellants	N_2O_4/50-50 or MMH	N_2O_4/50-50 or MMH	N_2O_4/MMH	N_2O_4/50-50 or MMH	
	Mixture Ratio (MR)	1.6		1.65	2.0 or 1.6	
	Expansion Ratio (ε)	47.5:1	60:1	90:1	40:1	
	Cooling	Ablation and Radiation (insulated)		Regenerative and Radiative (insulated)	Ablation and Radiation (insulated)	
	Burn Time (sec)	1000		2000	500	
	Engine Weight (lb)	389	12	481	315	
	Sized for and Listed	93 in x 58.4 in dia		85 in x 47.25 in dia	81 in x 47.5 in dia	
	Development Status:	In qual. for Apollo application. Requires Voyager qual.	Qualified for Apollo Requires Voyager qual.	Improved model in pre Phase II design Requires Voyager qual.	Qualified & flown in space at MR = 2.0 Requires Voyager qual.	
	Engine Reliability:	0.9989	0.9981			
	Minimum Impulse Bit:	1 m/sec	1 m/sec	1 m/sec	1 m/sec if shut down prior to full thrust	SELECTED:
	Engine Dispersions: Proportional to thrust level in pounds	9850 - good for injection 1050 - fair for trim	9850 - good for injection 100-400 - best for trim	17600 - worst for injection 960 - fair for trim	8000 - good for injection 8000 - worst for trim	LMDE

Physically, the Transtage and Agena engines are replaceable with the LMDE in the Voyager design concept because the LMDE has the largest overall envelope. Minor structural, dimensional, and plumbing changes would have to be made to accommodate the attachment points and propellant inlet connections that are different for each engine. The effect of the Agena engine gas generator and turbine exhaust on solar panel heating should be examined. It is probable that additional thermal insulation would be required for this engine. Additional plumbing and helium supply would also be required for the Agena engine, as it requires helium injection in the low thrust mode. The amount of additional helium and pressure bottle volume required has not been determined. It is estimated that an additional weight of 30 to 100 pounds might be required for that purpose.

None of the three candidate engines have been qualified to the Voyager mission requirements. All engines have been or will be developed for 50% N_2H_4/50% UDMH fuel. All manufacturers have stated that they could substitute MMH with little or no performance degradation. The Transtage engine has been qualified and flown in space at a mixture ratio of 2, and ground tested at a mixture ratio of 1.6. The Agena engine model used in this study is, at present, an advanced design pending Phase II development. The LMDE is currently in qualification and will be used in the Apollo program prior to its required use on Voyager.

3.6.4 Results

The result of the study was the selection of the LMDE engine for the Voyager propulsion subsystem. All candidate engines have advantageous features and all can be adapted to the Voyager application. The main criterion in selection of the LMDE is that its development status and its application on the Apollo program more closely support the Voyager propulsion subsystem application. The LMDE/C-1 engines were not used, despite a slight weight advantage, because of the added complexity to the flight spacecraft.

3.7 ORBITAL DATA RATE ANALYSIS

3.7.1 Purpose

The purpose of the study was to determine the required amount of data to be transmitted to Earth for the 1973 Mars Voyager orbital mission.

3.7.2 Scope

The data acquired by the spacecraft during the total orbital mission is composed of engineering, science, and relayed capsule data. Photoimaging by the science subsystem generates approximately 90% of the total orbital data. Therefore, this study is primarily involved in determining the amount of data generated by the imaging system. In determining this data quantity, consideration was given to the scientific objectives, to Lunar Orbiter results, and to practical spacecraft telecommunications limitations. The quantity of data generated by the imaging system is primarily a function of the resolution and surface coverage desired.

3.7.3 Analysis and Results

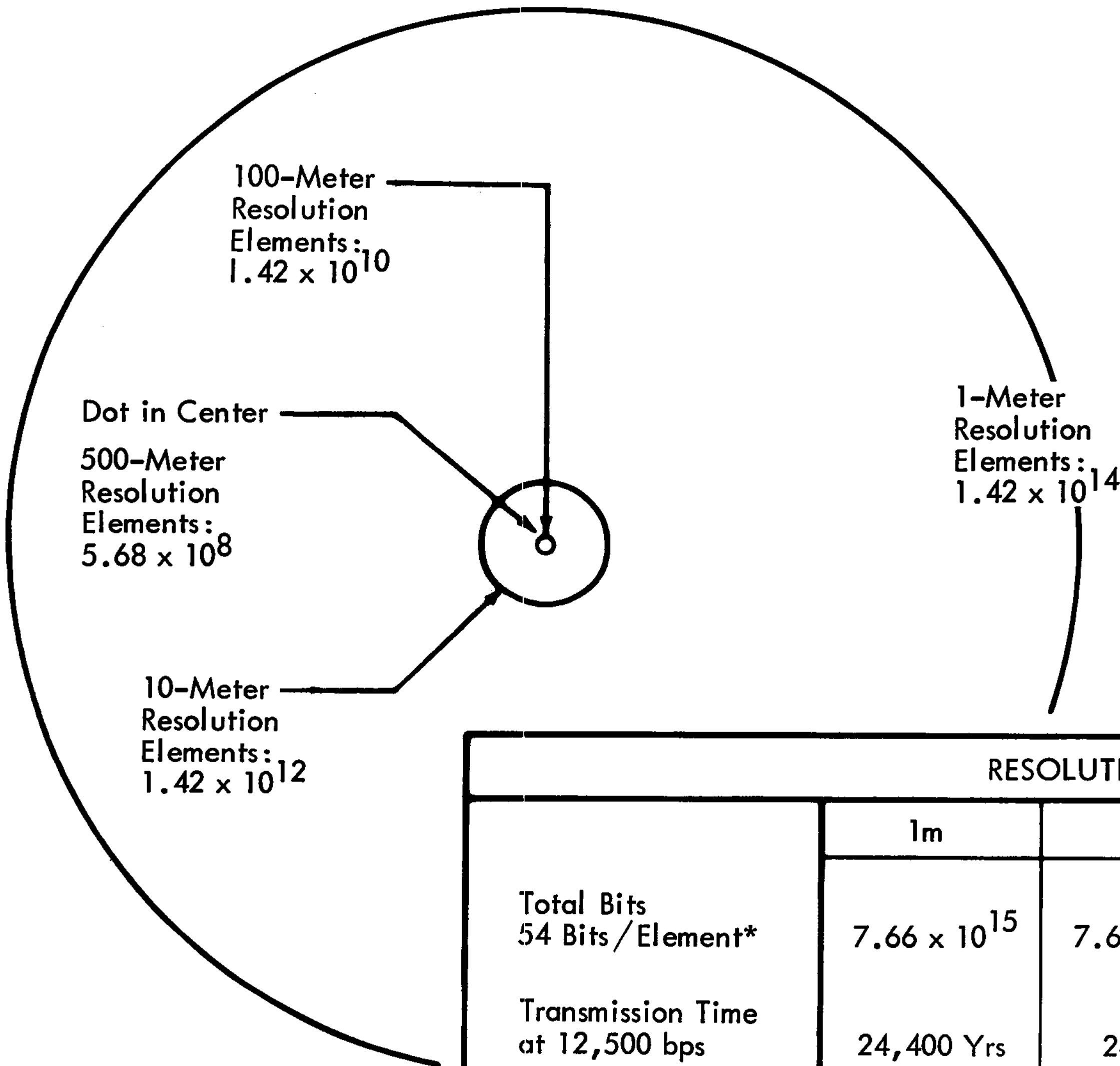
A summary of the photoimaging resolution and coverage achieved during the five Lunar Orbiter missions is given in Table 3.7-1. This program provided high resolution coverage of potential Apollo landing sites (Missions 1, 2, and 3) and a lower resolution, topographical survey of practically the total surface of the Moon (Missions 4 and 5).

Table 3.7-1: LUNAR ORBITER PHOTOGRAPHY RESOLUTION AND COVERAGE RESULTS

	MISSIONS 1, 2 & 3		MISSIONS 4 & 5	
RESOLUTION (METERS)	AREA (km^2)	LUNAR SURFACE (%)	AREA (km^2)	LUNAR SURFACE (%)
1	26,400	0.07	*	*
8	132,000	0.35	*	*
70 - 100	*	*	1.9×10^7	99.4%
1200 - 1600	*	*	1.9×10^7	

A similar approach seems applicable to the Voyager program. The higher resolution pictures would be taken to evaluate specific topographical features and assist in the search for extraterrestrial life on the planet. The lower resolution pictures will provide mapping of most of the surface of the planet to permit (1) selection of capsule landing sites, (2) identification of areas of scientific interest for more extensive examination, and (3) planning for subsequent missions.

The surface area of Mars is 1.42×10^8 km^2, approximately 7.5 times that of the moon. Figure 3.7-1 shows the quantity of data resulting from total coverage of Mars at various surface resolutions. One-meter resolution coverage of a Mars area comparable to Lunar Orbiter coverage of the moon (0.1%) would require 7.66×10^{12} bits and an attendant transmission time of 24.4 years at an average data rate of 12,500 bps. This transmission data bit rate is considered reasonable for the 1973 Voyager Mars mission, thereby indicating the impracticality of high resolution coverage of all of Mars. Therefore, a more reasonable compromise in the resolution and coverage selection is necessary within the constraints of practical data rates and a 6-month orbital mission. Only 80% of the orbital mission time is considered effective for data transmission accounting for Earth occultation and DSIF ground station switchover. A reasonable imaging selection is considered to be 0.1% high resolution coverage and coverage of a large fraction of the planet at medium resolution. Figure 3.7-2 presents four alternate imaging resolution/coverage combinations. The recommended science data quantity for the 1973 Mars Voyager mission is Alternate 2 in that figure. The selected approach provides imaging coverage of 0.1% of the planet's surface area at an average resolution of 10 meters, and 75% of the planet's surface area at an average resolution of 300 meters, corresponding to 1.41×10^{11} bits of data. The total science data for this imaging data acquisition mode is 1.57×10^{11} bits considering imaging data as 90% of the total science data. Since 80% of the mission time is assumed effective for data transmission, an average bit rate of 12,500 bps is required.



	RESOLUTION			
	1m	10m	100m	500m
Total Bits 54 Bits/Element*	7.66×10^{15}	7.66×10^{13}	7.66×10^{11}	3.03×10^{10}
Transmission Time at 12,500 bps	24,400 Yrs	244 Yrs	29.2 Mo.	35.2 Days

* Based on 6-bit gray scale and a Kell factor of 3.

Figure 3.7-1: IMAGING ANALYSIS

Alternate	High Resolution		Medium Resolution		Total Science Data (bits)	Average Data Rate (bps)
	Coverage of Planet %	Resolution* at Periapsis (meters)	Coverage of Planet %	Resolution* at Periapsis (meters)		
1	0.50	10	75	100	2.66×10^{11}	21,200
2	0.10	5	75	150	1.57×10^{11}	12,500
3	0.20	5	75	150	8.82×10^{10}	7,000
4	0.01	10	100	500	4.22×10^{10}	3,400

* Average resolution is twice value shown since imaging is taken from periapsis (500 km) to 3x periapsis (1500 km).

Figure 3.7-2: ALTERNATE IMAGING DATA OPTIONS

This selection (1) satisfies the 1973 Mars mission science objectives, (2) provides imaging coverage comparable to the Lunar Orbiter program, and (3) results in a data rate that is practical and within the current state of the art.

3.8 LASER TELECOMMUNICATION FEASIBILITY

3.8.1 Purpose

The purpose of this study was to conduct a gross feasibility evaluation of applicability of laser telecommunication to the 1973-1979 Mars Voyager missions.

3.8.2 Scope

The objectives of the Voyager missions to Mars during the 1970's include the acquisition of a large amount of imaging data of the planet's surface. The amount of the data returned to Earth from each mission, however, is limited by the RF telecommunication system planned for these time periods. Applying laser telecommunication, with its inherent higher transmission rate-capability, to these missions would enhance the achievement of the above objectives. Therefore, this study was conducted to determine the operational applicability and the development feasibility of a laser communications system for Voyager. No analysis was made of the use of a laser communications link for spacecraft tracking.

The study was conducted by first determining the current state of the art of lasers. This was accomplished by reviewing available technical material and consulting various other companies engaged in laser research. Using this material, a Voyager system concept was developed and its configuration evolution identified. A development program was outlined and a gross program schedule prepared. Care was taken to select only realistic data on the laser system to make a meaningful comparison with the more proven RF communication system capabilities.

In adapting the system to Voyager, care was taken to introduce laser communications into the Voyager mission in a gradual, sequential manner. This approach reduces peak program resource expenditures and makes maximum use of existing facilities.

3.8.3 System Concept

The laser telecommunication system concept consists of: (1) an Earth-based beacon transmitter telescope pointed towards Mars; (2) a spacecraft receiver/transmitter telescope which uses the Earth beacon signal for tracking purposes and transmits a laser data beam towards Earth; and (3) an Earth-based receiver for collecting the spacecraft laser data beam. Figure 3.8-1 illustrates this concept and Figure 3.8-2 identifies the primary functions for each of these major system elements.

Analyses of the laser transmitters included evaluation of both visible (argon-helium-neon) and middle infrared (carbon dioxide) lasers. The visible system was chosen for the 1975 mission because of its more advanced state of development. Based on consultation with laser research companies, significant improvements in visible gas laser efficiency and power output are expected to occur within

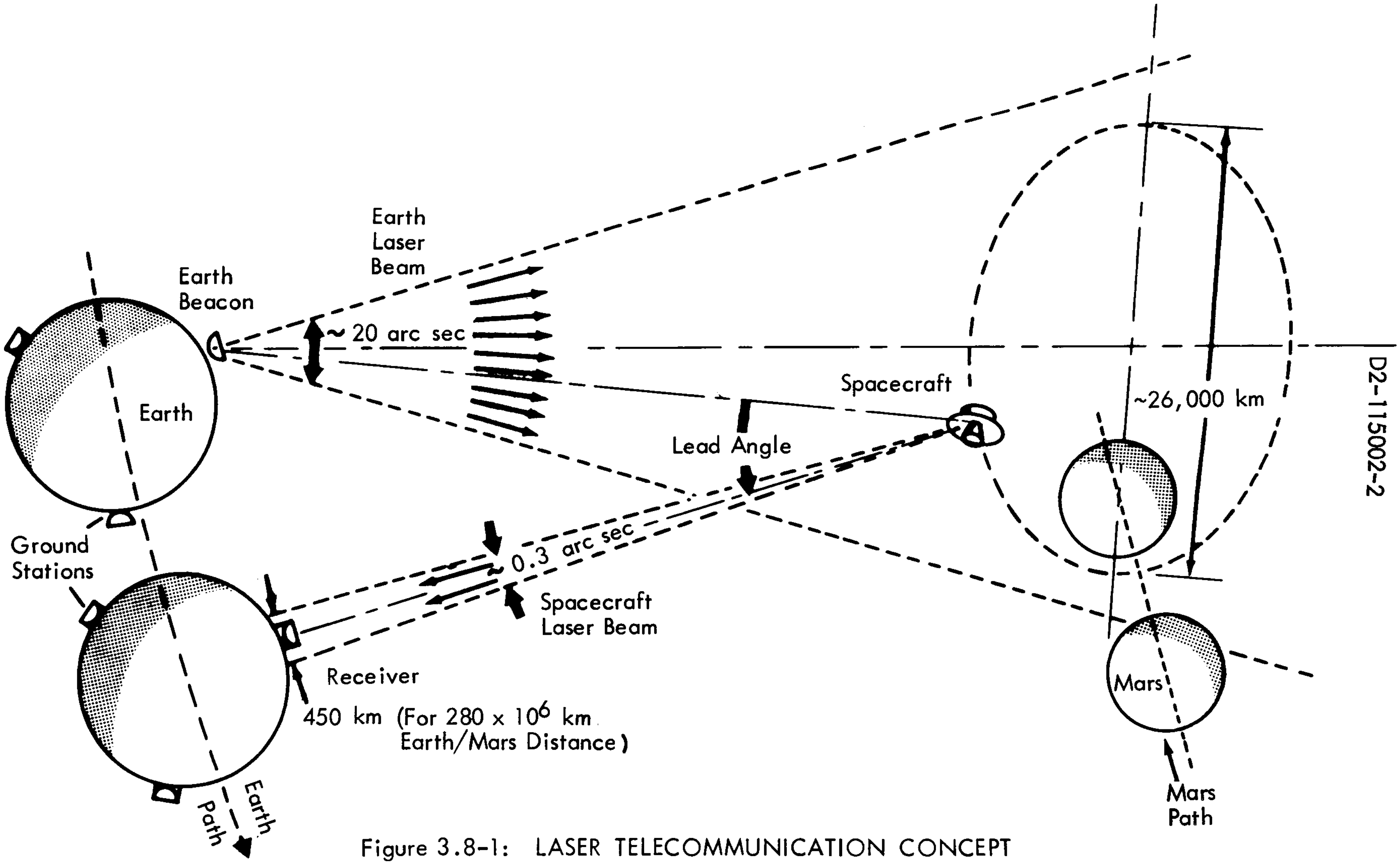


Figure 3.8-1: LASER TELECOMMUNICATION CONCEPT

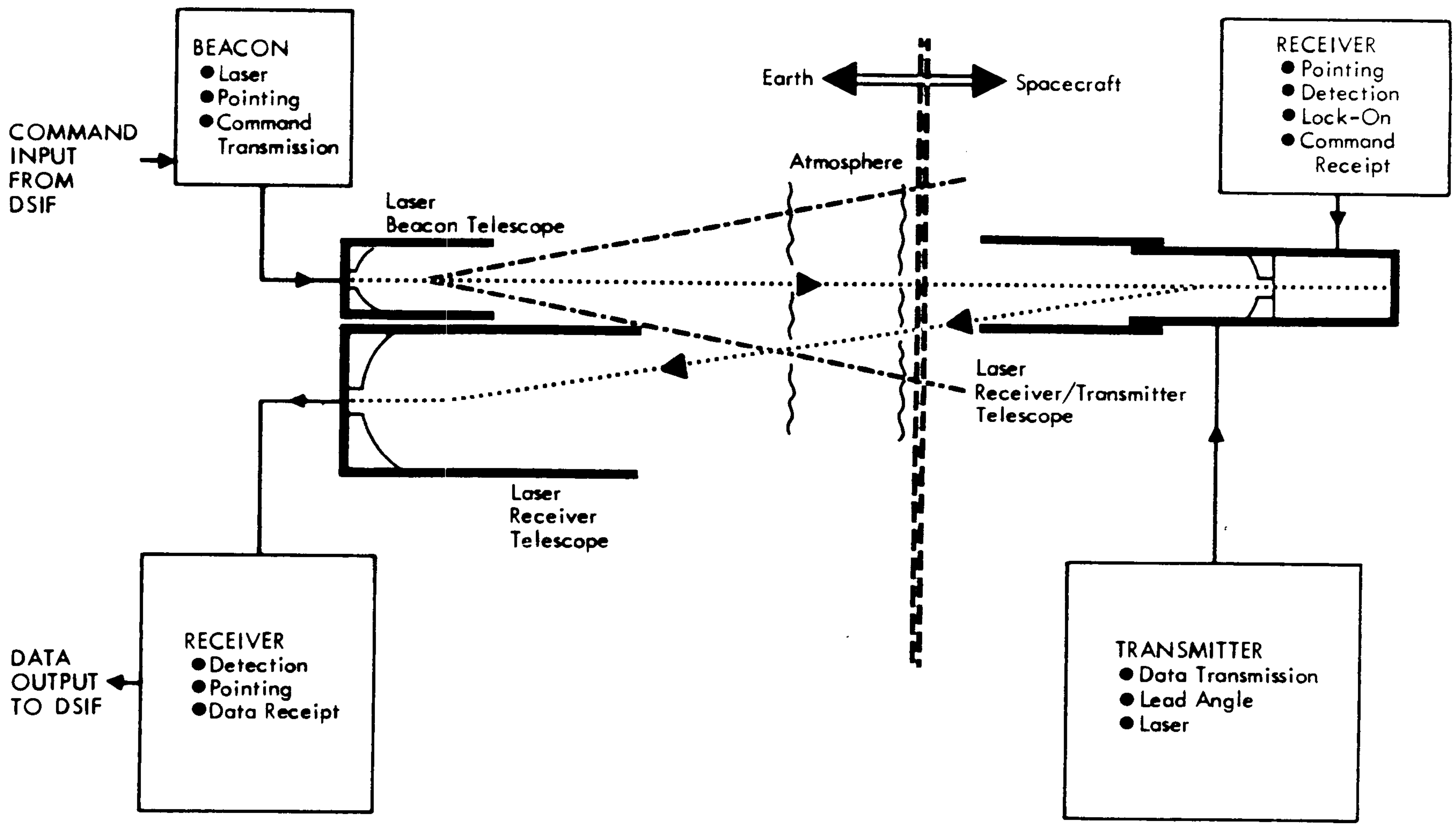


Figure 3.8-2: LASER TELECOMMUNICATION SYSTEM FUNCTIONAL CONCEPT

the next year. Conservative estimates of these improvements are reflected in the present study. Further improvements may increase capability and decrease the size of the configuration selected in this study.

The beamwidth of the Earth-based laser beacon beam can be relatively wide, since adequate ground power would be available. For this study, a visible frequency argon laser having a beamwidth of from 1 to 20 arc-seconds was analyzed. Beamwidths of this size appear to be necessary to account for refractive index fluctuations caused by the Earth's atmosphere. The 20 arc-second beamwidth is considered sufficient to illuminate the entire spacecraft orbit during its mission life. By contrast, the required spaceborne transmitter beamwidth is much narrower to give good ground-based detection for a reasonable spacecraft power level. In the selected concept, a beamwidth of 0.3 arc-second has been used. Therefore, spacecraft transmitter pointing accuracies of about 0.05 arc-second (rms) are required to maintain the beam on a chosen ground station. Since the spacecraft beam illuminates a small area on Earth, it must be pointed to Earth with a lead angle to compensate for the change in Earth position during the laser travel time.

The location of the ground station must be selected to minimize the overall degradation caused by the atmosphere, such as effects of absorption and scattering as well as intensity (scintillations) and angle of arrival fluctuations. The ground station will use a 10-meter effective aperture collector mirror. For this system, the quality of the collector need not be diffraction-limited. Preliminary studies have indicated that approximately 8 to 10 specifically located ground stations are required for continuous coverage to offset inclement weather conditions and the occultation of these ground-based stations due to rotation of the Earth. A fewer number of ground stations would be required if continuous coverage is not necessary.

The laser system will also be influenced by whether the Earth receiver is operating under daytime or nighttime Earth sky conditions. Because of the increased daytime background noise level, approximately a 10 times greater data rate can be achieved during Earth nighttime viewing at a given spacecraft power level. Also, a Sun-Earth-Mars angular limitation exists such that the ground receivers cannot adequately discriminate a spacecraft laser beam in close proximity to the Sun. For the lasers selected in this study, this angular limitation is estimated at 10 degrees. Data were developed to evaluate the laser day/night Sun angle limitations on each Voyager mission. The data are shown in Figure 3.8-3. The data show that the orbital portions of all four missions primarily occur during Earth-Mars daylight viewing. Therefore, the daylight laser bit rate is used for this study. The 10-degree Sun angle limitation will, in some missions, reduce the useful laser communication life by about 1-1/2 months.

A comparison was made of the data rates for the RF system used in the 1973 spacecraft baseline configuration and the laser system configuration concept developed in this study. The comparison is shown in Table 3.8-1. The laser capability is about seven times greater for an equal power input. It should be noted that the capability increases by a factor of 4 for the laser and by a factor of 2 for the RF as the spacecraft input power is increased. This difference in performance increase is caused by the light noise effect in the Earth's atmosphere becoming significantly less in comparison with the received signal as the laser power is increased.

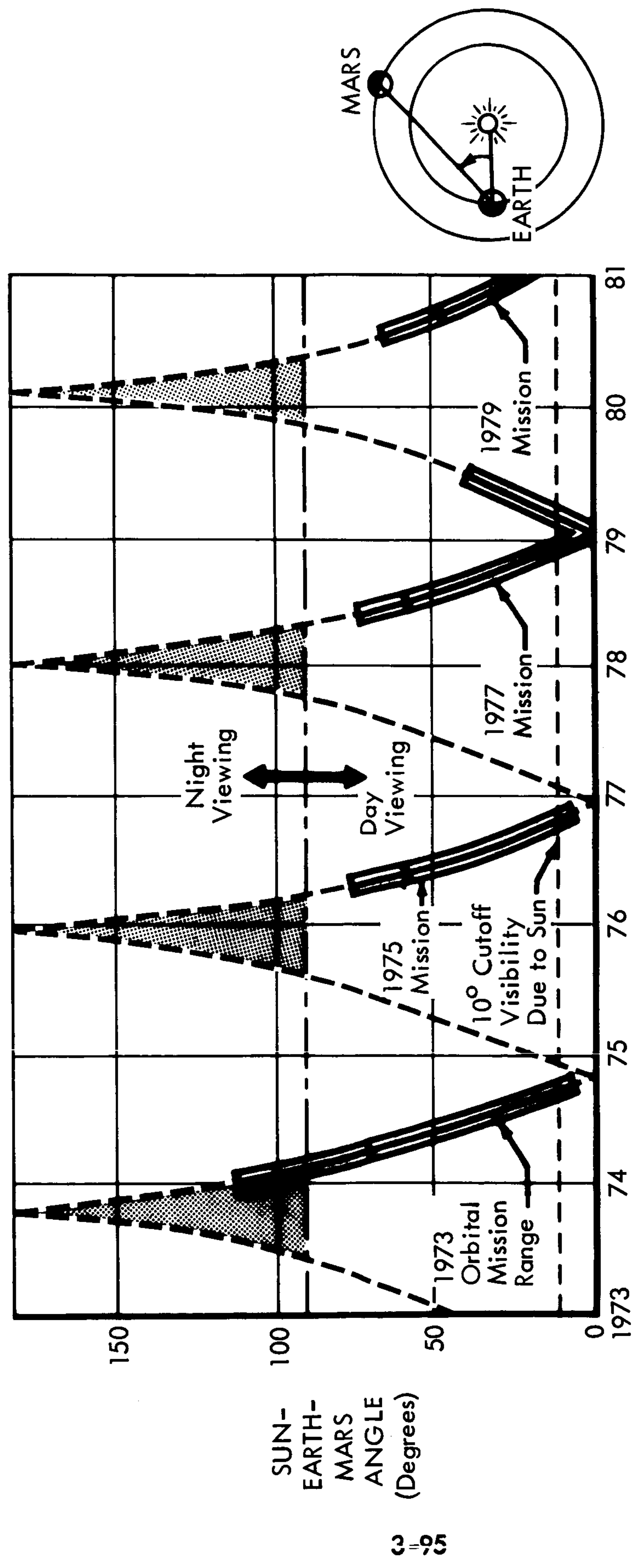


Figure 3.8-3: DAY/NIGHT AND SUN ANGLE EFFECTS ON LASER COMMUNICATIONS

Table 3.8-1: RF AND LASER PERFORMANCE COMPARISON

TYPE	SPACECRAFT INPUT POWER (Watts)	SPACECRAFT TRANSMITTER DIAMETER (Ft.)	DATA BIT RATE, (bps)	
			CD * (260×10^6 KM)	CD * (380×10^6 KM)
1973 Mission Baseline Configuration	150	13.7 (Antenna)	2.4×10^4	1.2×10^4
Laser	150	2 (Telescope)	2.4×10^5	5.5×10^4
Laser	300	2 (Telescope)	9.5×10^5	2.2×10^5

* CD = Communication distance.

The low gain portion of the spacecraft RF subsystem has omnidirectional capability for both receiving commands from Earth and transmitting low-rate engineering data back to Earth. This feature allows ground control during all spacecraft operations involving different spacecraft attitudes. Omnidirectional capability in the laser telecommunication system is not considered feasible. Therefore, a Voyager mission containing laser telecommunications should also include low gain RF omnidirectional communication equipment.

3.8.4 Evolution Plan

The plan for applying the laser telecommunication to Voyager calls for an initial installation on an experimental engineering basis, redundant to a full RF system. In subsequent missions, the dependency on laser transmission of data can be increased and the RF system decreased. Concurrent with the increased dependence on laser, the DSN facilities would gradually be expanded to include full laser capability. The laser system concept has the capability of transmitting commands from Earth to the spacecraft and the ability to track the spacecraft. However, the need for implementing laser tracking capability is not apparent, since the existing RF DSN has adequate tracking capability. The laser system evolution plan for this study is shown in Table 3.8-2. Laser command and tracking operation has been included in this table, but further studies may delete this requirement.

Table 3.8-2: LASER CONFIGURATION CONCEPT EVOLUTION PLAN

LINK	MISSION	1973		1975		1977 **		1979 **	
		LASER	RF BASELINE	LASER	RF BASELINE	LASER	RF* MINIMUM	LASER	RF* MINIMUM
Transmission to Earth	Primary		X		X	X		X	
	Backup			X (Experiment only)			X		X
Command	Primary		X		X		X		X
	Backup					X		X	
DSN Tracking			X		X		X	X	X
DSN Capability			Complete	Limited	Complete	Limited	Complete	Complete	Complete

*Low gain system only.

** Based on successful development of the 1975 experiment

3.8.5 Configuration

Significant sizing values for key system elements have been identified as shown in Table 3.8-3.

Table 3.8-3: LASER SYSTEM CONFIGURATION DESCRIPTION

LASER SYSTEM ELEMENTS	TELESCOPE SIZE DIAMETER (ft)	TELESCOPE SIZE LENGTH (ft)	OPTICAL SYSTEM FINISH (RMS)	POINTING ACCURACY (arc sec)	LASER GAS
Ground Beacon	2	7	$\lambda/5$ to $\lambda/10$	± 1	Argon
Ground Receiver	33	100	2λ	± 1	——
Spacecraft Receiver/Transmitter Telescope	2	7.5 (11.5 with Extended Sun Shade)	$\lambda/40$ (Diffraction Limited)	± 0.05	Helium -Neon

λ = Wavelength of light at 5000 angstroms.

The ground station beacon and receiver based on these values are considered to be within today's state of the art. Therefore, no further effort on detailing the configuration of this ground equipment was conducted for this study. The spacecraft receiver-transmitter portion of the system is considered a significant development task. This observation is illustrated by the requirements for defraction-limited optics and $\pm$ .05 arc-second pointing accuracy of the spacecraft laser telescope.

The spacecraft receiver/transmitter is primarily a 4-foot long, 24-inch clear aperture Ritchey-Chretien telescope with a 4-foot sunshield extension and an equipment section extending the overall length to 11.5 feet as shown in Figure 3.8-4. The optical laser schematic of the telescope is also shown in the figure. A key aspect of the schematic is the coarse, medium and fine pointing concepts of the system. The coarse pointing may be provided by mounting the telescope assembly on the gimbaled RF support. In this manner, the laser telescope will be initially pointed to Earth to the same accuracy as the RF antenna. The medium pointing mode is accomplished by using an optical sensor within the telescope. The signal from this sensor is used to operate an electrical/mechanical drive mechanism located on a gimbal mount to which the telescope is mounted. This mechanism accurately aligns the telescope assembly to Earth. The fine guidance pointing operation occurs within the telescope by using an optical/electrical beam splitting mechanism and by moving an optical lens with microactuators. The specific values for each of the pointing modes are shown in Figure 3.8-5. This pointing operation

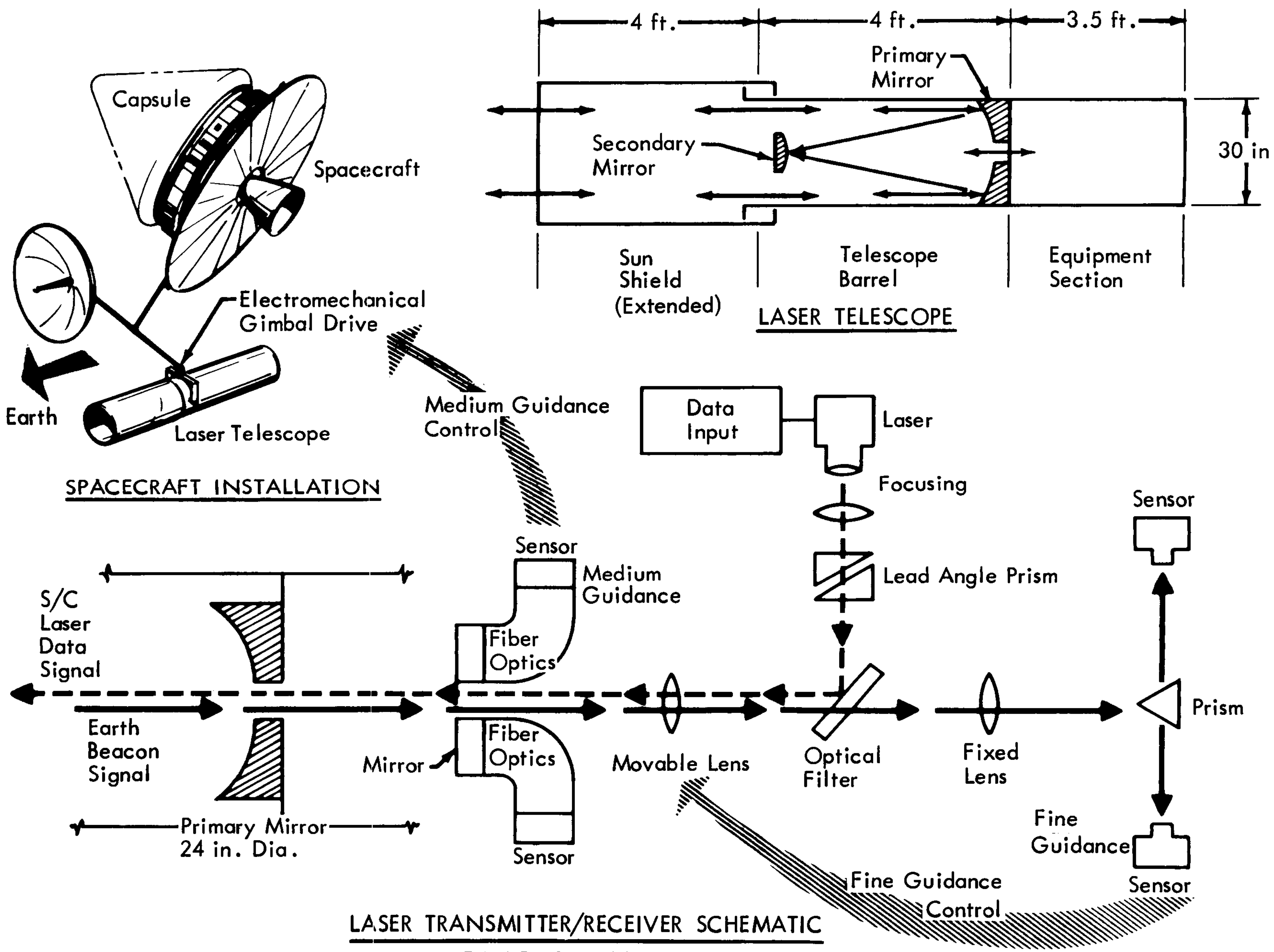
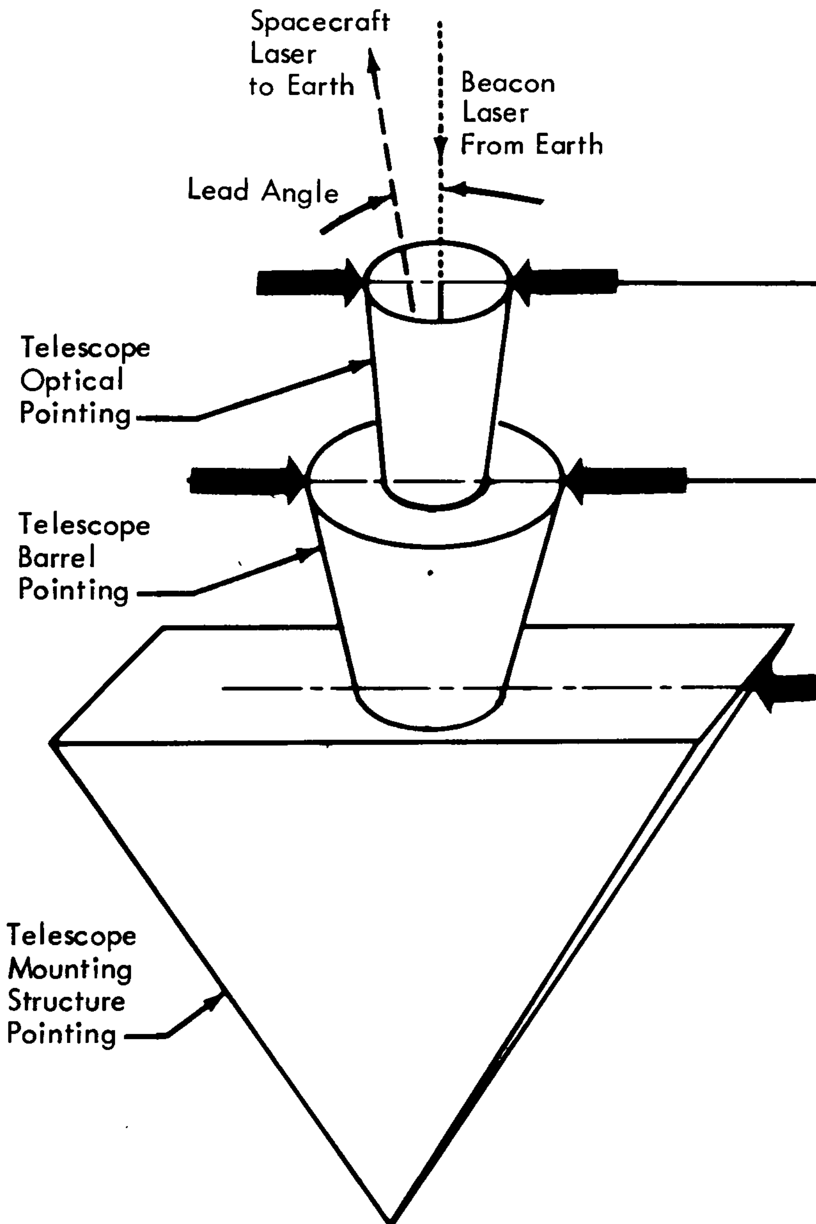


Figure 3.8-4: SPACECRAFT LASER CONFIGURATION CONCEPT



MODE	POINTING		SENSING		MOVING ELEMENT
	Range	Accuracy	Signal	Location	
FINE	2 minutes	± 0.05 arc sec	Earth Laser Beacon	Telescope	Telescope Optics
MEDIUM	2 degrees	± 1.0 arc sec	Earth Laser Beacon	Telescope	Telescope Assembly
COARSE	280 by 200 degrees	± 1 deg	Sun Canopus	Space-craft	Telescope Mounting Structure

Figure 3.8-5: SPACECRAFT LASER POINTING CONCEPT

is used to establish a precise spacecraft-to-Earth reference line from which the spacecraft laser signal can be accurately aligned for transmission to Earth. The spacecraft laser signal is sent to Earth through the same optics simultaneously while receiving the Earth signal. The lead angle required in pointing the beam to intercept Earth is accomplished by moving an optical prism element. This angle can either be calculated aboard the spacecraft or on the ground and transmitted to the spacecraft via the command link.

The configuration concept for using the laser system for command and tracking purposes has not been illustrated. Should this feature be used, the sending of commands to the spacecraft would be accomplished by coding the ground beacon signal and having decoding equipment aboard the spacecraft similar to the existing RF decoding equipment. Tracking by the laser system could be accomplished in a manner similar to the method used by the existing DSN RF system.

For the 1975 mission when the laser system is used as an experiment, no additional spacecraft power provisions are planned. With today's spacecraft design there will be periods in the orbit when 150 watts of power will be available. During these periods the laser will be used. In subsequent missions when the RF high gain antenna system is removed, the power it previously used can be used by the laser. In these later missions more spacecraft power for the laser may be necessary to fully utilize the laser telecommunication systems potential. The weight of a 1975 Voyager spacecraft equipped with a laser is expected to increase approximately 700 pounds. This increase includes both the weight of the laser subsystem (telescope, laser source, electronics pointing mechanism, mounting structure, etc.) and the increase in propellant to accommodate the weight change. Sufficient launch vehicle payload capability exists to absorb this weight increase.

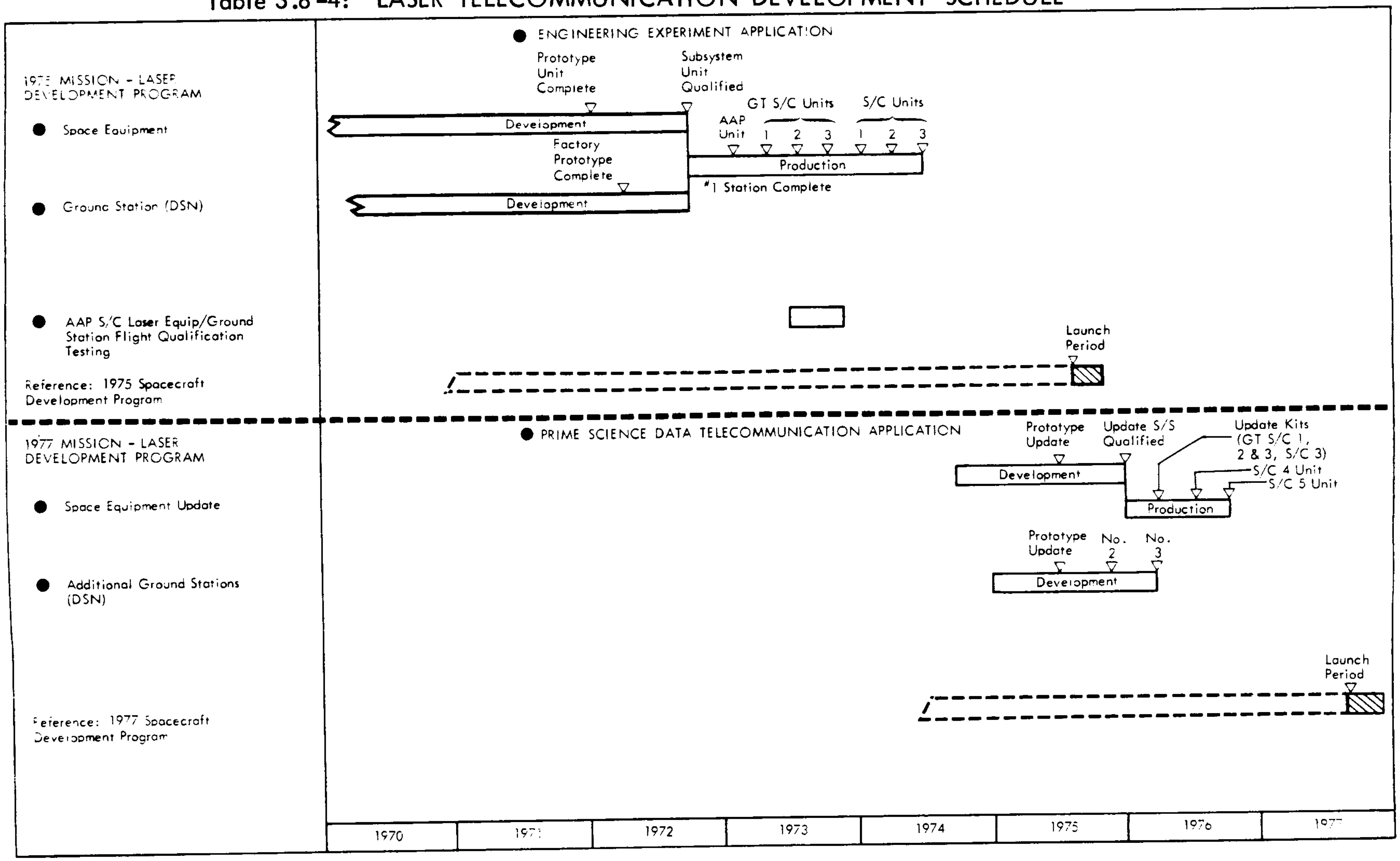
Thermal design considerations must recognize the low efficiency of the laser-producing equipment (less than 1%). This loss in power is converted to heat which must be dissipated in a manner that does not affect the alignment of the laser subsystem optics.

3.8.6 Development Plan

The development plan established in this study is based on the previously presented evolution plan. The basic laser equipment for the ground stations and spacecraft would be developed as an experiment for the 1975 mission. Prototype units of each would be used in a system-type demonstration and evaluation test program. After these tests and the ground qualification of the spacecraft equipment, a unit will be used in an Earth-orbit Apollo Application Program (AAP) mission (preferably in a near synchronous orbit to reduce ground tracking rate). Subsequent spacecraft units would be produced to support the Voyager spacecraft ground testing program and to provide the units required for flight. In this plan, only one ground station has been considered for the 1975 mission experiment. Further laser ground system development is considered necessary to support the 1977 mission should the laser equipment development prove satisfactory.

A schedule for the laser system development is shown in Table 3.8-4 which was based on the current spacecraft program schedule. It is estimated that nine spacecraft laser subsystem units will be required for the 1975 experiment.

Table 3.8-4: LASER TELECOMMUNICATION DEVELOPMENT SCHEDULE



3.8.7 Major Development Problems

Certain major development problems can be anticipated for the laser communication system. These are:

1) Development of the spacecraft optical equipment to provide the required accuracies after having been subjected to the launch vehicle and spacecraft propulsion subsystem induced environments and when operating in space.

2) Development of an electromechanical space-qualified mechanism required to point the laser telescope assembly to the required accuracy ($\pm$ 10 arc-seconds).

3) Development of a thermal design that will thermally control both the telescope and laser optics to the degree required.

4) Development of a method for establishing spacecraft telescope coordinates with respect to the Earth, for orienting the spacecraft laser lead angle to the required accuracy (about $\pm$ 1 arc-second).

5) Determination of atmosphere propagation characteristics on laser beams for use in designing the ground station. The laser communications satellite experiment (LCSE) program may be used to obtain this propagation data.

3.8.8 Results

Three major results of the study are:

1) Laser data transmission capabilities identified in this study are significant in providing the spacecraft with greater capability in meeting mission objectives. Therefore, further study effort is justified to define more fully the system and the related development program so a project decision can be made.

2) An experimental laser telecommunication system is considered feasible for the 1975 Mars mission.

3) Further studies should include an evaluation of laser communication use for planets other than Mars. This is considered necessary to ensure that the development of ground stations and spacecraft equipment for Mars will have future application to other space exploration programs.

3.9 RF HIGH DATA RATE STUDY

3.9.1 Purpose

The purpose of this study was to explore the feasibility of increasing the information bandwidth of the spacecraft-to-Earth data link in the post-1973 missions.

3.9.2 Scope

The evolution of the orbital science payload will significantly increase the amount of information to be transmitted from the spacecraft to Earth. This is true in particular for the photoimaging experiment for which a large film camera is postulated in 1977 and 1979. The development of laser telecommunications offers one potential solution to the problem of increasing the data rate. However, the resources required to develop a new mission operation system (MOS) for lasers are large. Hence, an alternate solution is the further development of the present RF telecommunications system which uses an existing MOS (i.e., the DSN). This study is concerned with such further developments of the RF system.

Two approaches were considered and their implications examined:

1) Increasing the spacecraft effective radiated power (ERP) by the use of higher output power amplifiers and by incorporating a higher gain antenna.

2) Increasing the efficiency of the utilization of the received signal power through different modulation techniques.

The study evolved in three steps, as follows:

1) Trades among the spacecraft RF parameters -- Spacecraft ERP was derived as a function of antenna size for various spacecraft pointing capabilities.

2) Link performance -- Based on the characteristics of the DSIF receiving equipment, the received carrier-to-noise density ratio (C/kT_S) was obtained as a function of the communications range and of the spacecraft ERP.

3) Trades among modulation-demodulation techniques -- The information (video) bandwidth, over which a satisfactory output signal-to-noise ratio can be maintained, was obtained as a function of the C/kT_S for various modulation/demodulation techniques. Digital and analog modulation techniques were compared and the effect of preemphasis and deemphasis was considered. Modifications required of both spacecraft and ground equipment were determined for each modulation scheme.

3.9.3 Analysis and Results

3.9.3.1 RF Parameters

Effective radiated power (ERP) constitutes the primary criterion for comparing the performance of various spacecraft communications systems. ERP is defined as the db sum of the transmitter power and the net antenna gain, less the RF losses in the spacecraft. The transmitter power is obtained from the nominal output power of the final amplifier.

The net gain of the transmitting antenna is given by the on-axis gain of the antenna, reduced by (1) the antenna gain tolerance and (2) the antenna pointing loss. This pointing loss is the degradation of the antenna gain caused by the antenna not being boresighted exactly on the receiving station.

RF losses comprise the transmitter output tolerance (worst-case output, including effects of aging) and the worst-case value of the transmitting circuit losses.

Output Power -- Tube manufacturers list S-band traveling wave tube (TWT) amplifiers in the 50- to 100-watt range (WJ 274-1, EM 1260) and electrostatically focused klystron (ESFK) amplifiers from 100 watts to 500 watts (Litton L-5044, Eimac 3065A). Consequently, four nominal transmitter output levels were considered: 50 watts, 100 watts, 200 watts, and 500 watts.

Antenna Gain -- For the assumed parabolic-reflector type antenna, the nominal on-axis gain is a function of (1) the antenna efficiency, (2) the antenna diameter, and (3) the transmission wavelength. Assuming a 50% antenna efficiency (worst-case value) and using an operating frequency of 2295 mHz, the net gain (G_o) of the antenna becomes

$$G_o = 14.3 + 20 \log D$$

where D is the antenna diameter in feet. Antenna diameters between 2 feet and 20 feet were considered in this analysis. Antenna gain tolerance should include effects of antenna illumination efficiency and of reflector surface roughness. Using the worst-case antenna efficiency and a minimum surface roughness of 0.1-inch, the antenna gain tolerance equals zero db.

Pointing Loss -- The pointing loss depends on the shape of the antenna beam and on the pointing accuracy. The pointing loss, L_p, is related to the cone angle, θ_C (i.e., the angle between the boresight axis of the antenna and the direction of the transmission) by

$$L_p = 12 \, (\theta_C/\theta_{HP})^2$$

where L_p is measured in db and θ_{HP} is the half-power beamwidth. The per-axis pointing requirement is the rectangular component of θ_C. It becomes a maximum at a value of 0.707 θ_C. Previous design studies assumed per-axis pointing requirements of 0.60 degree (practical) and 0.93 degree (conservative).

RF Losses -- In general, the spacecraft RF losses vary with the design of the communications subsystem, including the extent of the redundancy switching provisions. The values of the RF losses determined in previous studies (i.e., Task B) were used for analyzing all communication system configurations. The spacecraft RF losses used in this analysis therefore consisted of:

Transmitting circuit loss	3.1 db
Transmitter output tolerance (including effects of aging)	1.1 db
Total	4.2 db

Spacecraft Constraints -- The selection of the transmitter output power is influenced by the capabilities of the power and thermal control subsystems. Manufacturers' data for the tubes listed above indicate that the efficiency of the RF amplifiers ranges from 30 to 50%. The higher efficiency value is usually obtained at the expense of other performance characteristics. A value of 30% (which includes the power supply efficiency) therefore was used for this study.

Based on this efficiency, the power and thermal control subsystem design establish an upper limit of the allowable RF output power. The maximum nominal RF output power meriting serious consideration appears to be 150 watts.

An increase of the antenna diameter reduces the allowable pointing error. Three values of the maximum pointing error were considered in this study:

1) $\pm$ 0.93 degree -- conservative

2) $\pm$ 0.60 degree -- obtainable with ground calibration of attitude control sensors and antenna alignment (practical)

3) + 0.35 degree -- obtainable with inflight calibration of attitude control (A/C) sensors and antenna alignment (feasible).

The table below shows a breakdown of the above pointing errors.

Breakdown of Pointing Error Components

	Conservative	Practical	Feasible
Per-axis pointing error	$\pm$ 0.93 deg	$\pm$ 0.60 deg	$\pm$ 0.35 deg
Cone angle	$\pm$ 1.30 deg	$\pm$ 0.85 deg	$\pm$ 0.50 deg
Error Components			
A/C Limit Cycle	$\pm$ 0.40 deg	$\pm$ 0.30 deg	$\pm$ 0.20 deg
A/C Reference Sensor*	$\pm$ 0.15 deg	$\pm$ 0.15 deg	$\pm$ 0.05 deg
Structure*	$\pm$ 0.47 deg	$\pm$ 0.24 deg	$\pm$ 0.10 deg
Antenna Control*	$\pm$ 0.18 deg	$\pm$ 0.10 deg	$\pm$ 0.10 deg

*These components must be root-sum-squared.

To avoid an excessive increase of polarization loss, the pointing error is restricted to the half-power beamwidth, i.e., $L_p \leq 3$ db.

Antenna Size versus Effective Radiated Power (ERP) -- Using the above results, Figure 3.9-1 shows the trade between antenna diameter and spacecraft ERP for pointing errors of 0, $\pm$ 0.35, $\pm$ 0.60, and $\pm$ 0.93 degree. These curves indicate the diminishing trend of achieving ERP increases through an increase of antenna size.

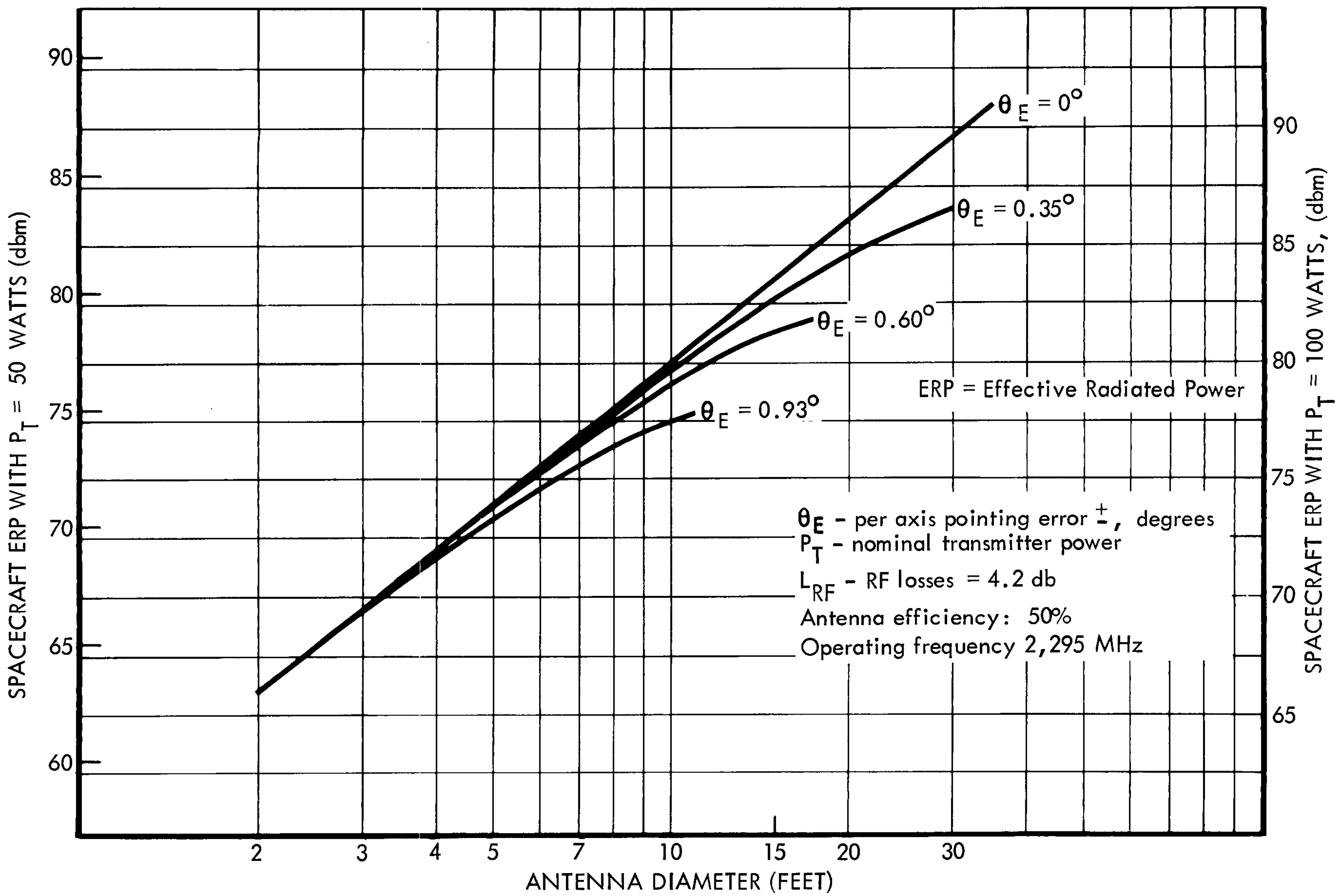


Figure 3.9-1: SPACECRAFT ERP VS. ANTENNA DIAMETER (CONSTANT POINTING ERROR)

3.9.3.2 Link Performance

The received signal power is a function of (1) the spacecraft ERP, (2) the communications distance, and (3) the parameters of the receiving system.

The receiving system parameters used in this study were the worst-case values shown below.

Receiving System Parameters

Parameter	Value
Receiving antenna gain (210-ft dia.)	60.9 db
Receiving antenna pointing loss	-0.2 db
Polarization loss	-0.1 db
Receiving system noise temperature, T_s	35° K

The received carrier power (C) is the db sum of (1) the spacecraft ERP, (2) the receiving antenna gain, (3) the receiving antenna pointing loss, (4) the polarization loss, and (5) the free space loss. The free space loss is the signal attenuation with range, caused by spreading. This loss is a function of range (R) and transmission frequency. With the values chosen above, the received carrier becomes

$$C = (ERP) - 20 \log R - 39.1$$

The received noise power density is given by the product of the Boltzmann's constant K and the system noise temperature T_s. In db terms, k = -198.6 dbm/Hz, and T_s = 15.4 db at 35° K, so that the noise power density, kT_s = -183.2 dbm/Hz. Thus, the received carrier-to-noise power density ratio $(C/kT_s)_{db}$ = (ERP) - 20 log R + 144.1.

Figure 3.9-2 shows the variation of C/kT_s with the spacecraft ERP for selected values of range.

3.9.3.3 Comparison of Modulation Techniques

The two major classes of modulation techniques consist of either analog or digital methods. Each class has several variations. Digital systems can be coded or uncoded. Analog systems can use different combinations of phase, frequency or amplitude modulation.

Digital techniques are more amenable to systems having requirements for multiple data rates and the multiplexing of many different data sources. In addition, when data are recorded prior to transmission, signal fidelity is easier to maintain in a digital format. Thus, digital telemetry systems have been the historical choice for deep space missions and early Voyager requirements specified a digital interface between the science payload and the data transmission equipment. Analog techniques

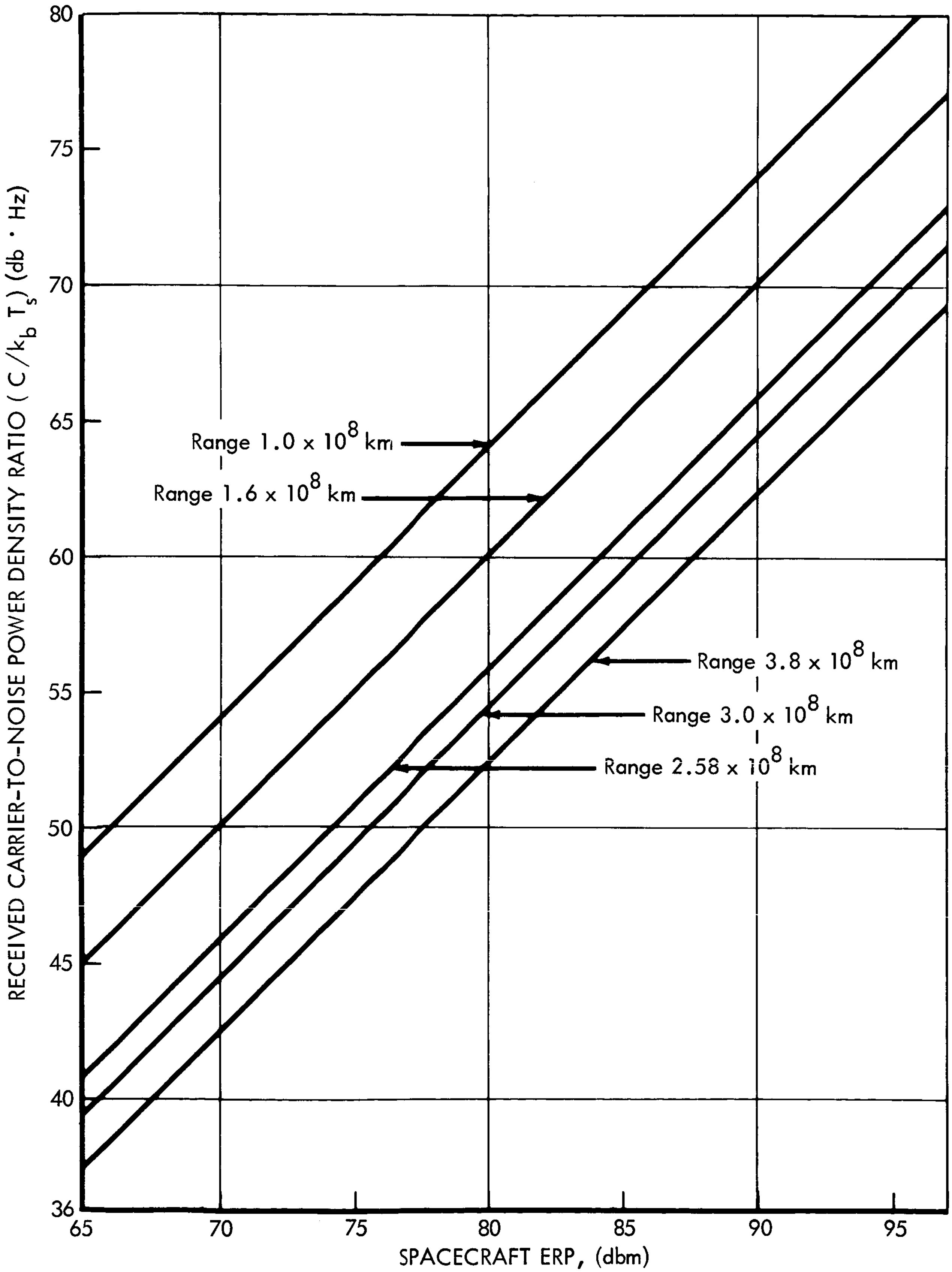


Figure 3.9-2: SPACECRAFT-TO-EARTH LINK PERFORMANCE

however, offer a potential improvement in modulation efficiency and merit consideration when the payload is predominantly a single analog sensor (e.g., a higher resolution film camera) with a high information content.

Digital Modulation Techniques -- Two digital modulation techniques were considered: uncoded phase-shift-keying/phase modulation (PSK/PM) and biorthogonal (16,5) coded PSK/PM.

In both techniques, a digital bitstream shifts the phase of a subcarrier $\pm$ 90 degrees. The resulting phase-shift-keyed subcarrier, in turn, modulates the phase of the RF carrier. The two techniques above differ in (1) the generation of the modulating bitstream and (2) the bit detection processes. In both techniques, the original analog data is sampled and quantized to 64 (2^6) levels. Thus, each data sample results in a 6-bit pulse code modulation (PCM) word. This PCM bitstream constitutes the modulating signal for the uncoded technique. In the coded technique, every 5-bit block of the PCM bitstream is further encoded in a 16-bit transmission word. In the demodulation process, the block-coded signal is detected by cross-correlating each received 16-bit block against the 32 signals representing the 5-bit blocks of the original PCM bitstream.

The required probability of a bit detection error is less than 0.005. The following table shows the theoretical and practical bit-energy-to-noise power density ratios necessary to achieve this performance level.

Required Bit-Energy-to-Noise Power Density Ratios (E/kT_S)
Bit Error Probability $\leq$ 0.005)

Modulation Technique	E/kT_s (Theoretical)	E/kT_s (Practical)
Uncoded coherent PSK	5.3	7.8
(16,5) coded PSK	3.5	6.0

The practical values in the above table include a 2.5-db allowance for worst-case effects of the demodulating hardware. Figure 3.9-3 shows the bit rate as a function of carrier-to-noise density ratio, C/kT_S, for both coded and uncoded digital modulation techniques, assuming a demodulation loss of 3.6 db.

Analog Modulation Techniques -- Four analog modulation techniques are considered: (1) FM/PM, (2) VSB-AM/PM, (3) PM, and (4) FM. With the first two modulation techniques, the video information modulates a subcarrier which, in turn, phase-modulates the carrier. With these techniques, an additional channel for engineering data transmission can be accommodated. FM/PM modulation allows tracking of the doppler shift simultaneously with the video readout. In the VSB AM/PM modulation technique, the video transmission must be interrupted to obtain doppler data. With the PM and FM techniques, on the other hand, the video signal modulates the **carrier directly. Therefore, a second transponder is required to handle the turnaround transmission functions--doppler tracking and ranging--and to provide an engineering data channel.**

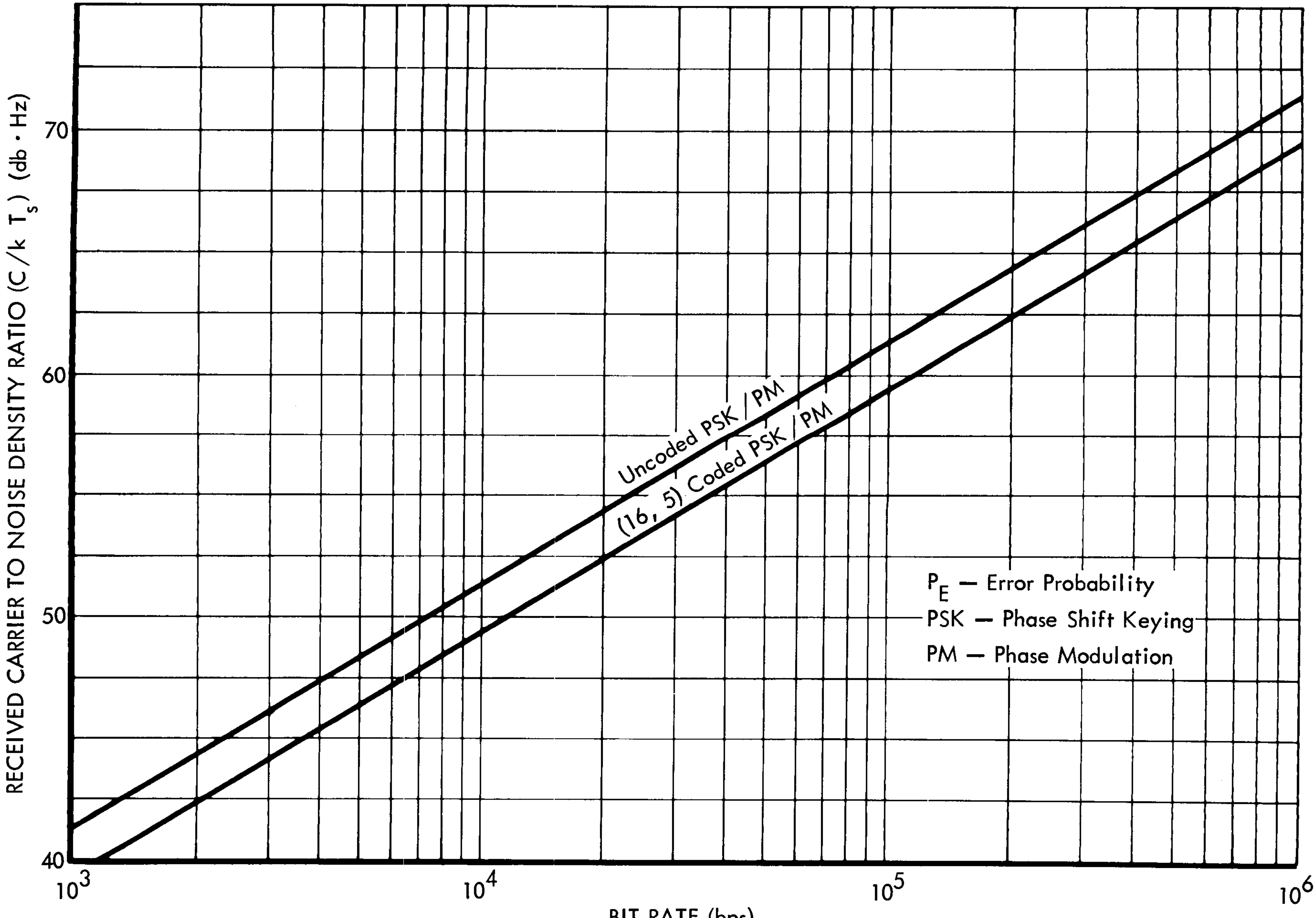


Figure 3.9-3: BIT-RATE CAPABILITY OF DIGITAL MODULATION TECHNIQUES
$P_E \leq 0.005$

The video bandwidths of the four modulation techniques were compared as a function of the carrier-to-noise power density ratio (C/kT_S). For this comparison, the following two constraints applied:

1) Output signal-to-noise ratio is equal to or greater than a specified value.

2) Carrier-to-noise ratio is equal to or greater than a specified value (carrier-to-noise threshold).

The output signal-to-noise ratio $(S/N)_o$ for all four of the modulation techniques is a function of (1) a constant (M) characterizing the modulation techniques, (2) the modulation index (m), (3) the modulation loss (L_m), (4) the carrier-to-noise density ratio (C/kT_S), (5) deemphasis improvement (D), and (6) the maximum video frequency (fm).

The table below shows the db values of the known parameters.

Modulation Technique	M	D: No Deemphasis	D: 18 db Deemphasis	Lm
FM/PM	4.8 db	0 db	14.2 db	-3.6 db
VSB-AM/PM	-1.0 db	0 db	5.8 db	0 db
PM	0 db	0 db	7.4 db	0 db
FM	4.8 db	0 db	14.2 db	0 db

The analog modulation techniques were compared on the basis of a 23.5-db (rms/rms) output-signal-to-noise ratio, which is the Lunar Orbiter design value.

Based on the required output signal-to-noise ratio constraint, the relationship between the maximum video frequency (fm) and carrier-to-noise density ratio (C/kT_s) is found to be:

$$10 \log fm = M + D - (S/N)_o + 10 \log \left(\tfrac{1}{2} m^2\right) + L_m + C/kT_s$$

Another relationship between the same two quantities arises from the required threshold carrier-to-noise ratio. The carrier-to-noise ratio $\left(\frac{C}{N}\right)$ is a function of Lm, C/kT_s, and B_{if}, where B_{if}, the IF bandwidth, is a function of the modulation index (m), $\frac{C}{N} = L_m + \frac{C}{kT_s} - 10 \text{ Log } B_{if}$. The available data bandwidths of the four analog modulation techniques are compared in Figure 3.9-4 assuming no deemphasis and assuming an 18-db deemphasis.

Modulation Considerations -- On the basis of the preceding data, Figure 3.9-4 indicates a clear-cut superiority of the FM carrier modulation technique. However, a number of other factors must be considered in the final selection of the modulation technique.

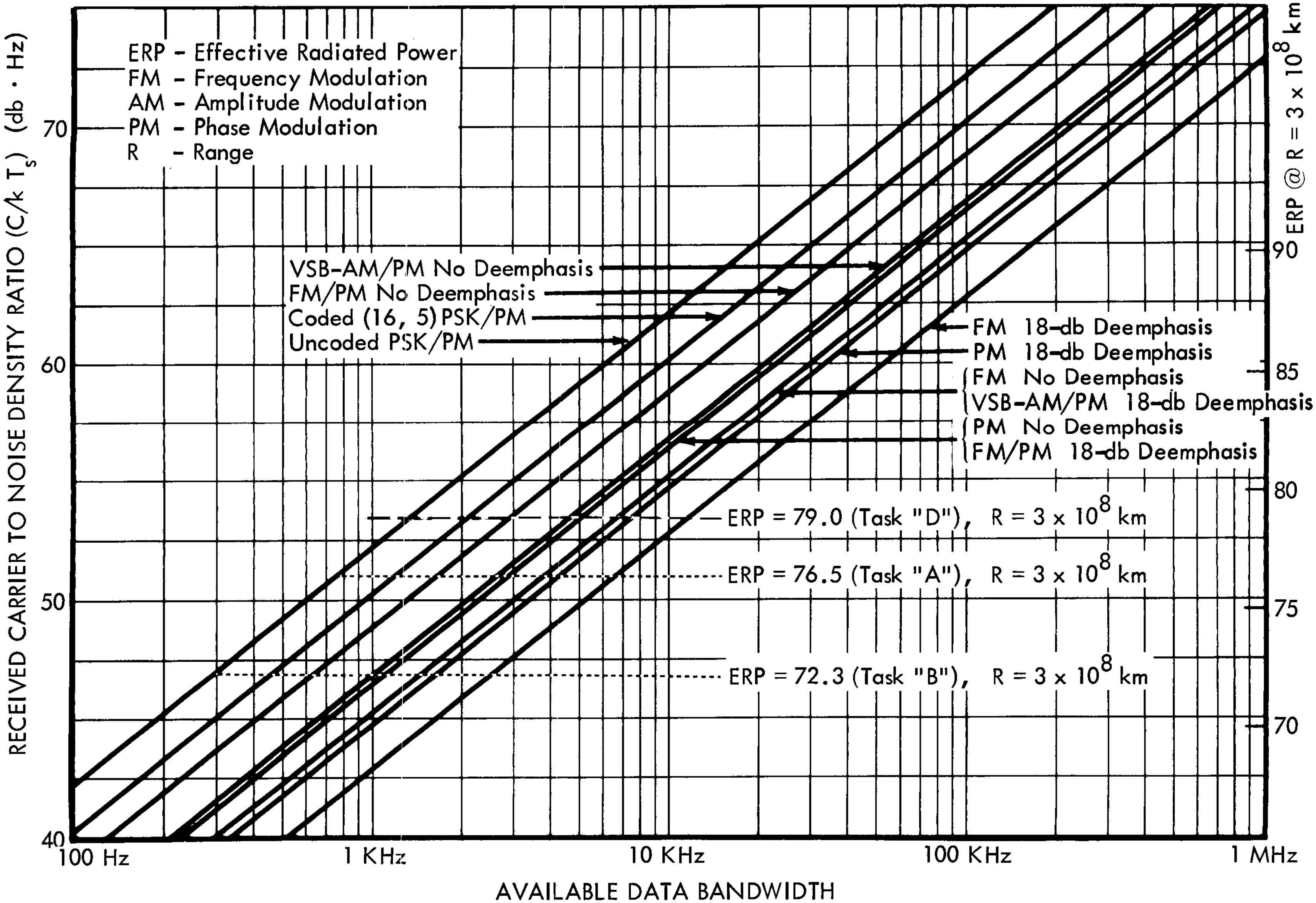


Figure 3.9-4: COMPARISON OF THE BANDWIDTH CAPABILITIES OF ANALOG AND DIGITAL MODULATION TECHNIQUES

1) The digital techniques, in general, provide a superior performance if the signal is to be relayed, either in real or nonreal time. This aspect becomes important if the signal received at a remote ground station will be recorded for a subsequent processing at a central site.

2) The digital techniques lend themselves easier to multiple data rate capability.

3) The carrier modulation techniques, FM and PM, have a poor low frequency response because of carrier instabilities (doppler and thermal drift). In general, TV systems require good low frequency response down to the field scan rate, while a photo readout system will operate satisfactorily if the low frequency response encompasses the line scan rate. A satisfactory operation of this modulation technique thus requires that the line scan frequency is well above the carrier frequency uncertainty.

4) The VSB-AM/PM modulation technique was evaluated by a direct scaling of the Lunar Orbiter communications subsystem. If this technique is adopted, a more detailed study of the actual filter requirements will be necessary.

5) The FM/PM system can be used as a single-subcarrier video transmission link. Omission of the second subcarrier reduces the modulation loss, resulting in a 1 to 1.5-db improvement over the two-channel configuration. As with the FM and PM techniques, a second link is then necessary to provide the other communications functions.

6) The amount of pre-and deemphasis available depends on the frequency response of the signal source. The 18-db preemphasis can be obtained with a Lunar-Orbiter-type film readout system. With a slow-scan vidicon readout, the available preemphasis will be about 6 to 9 db.

3.9.4 Conclusions

The link analysis of Section 1.2.3, Volume III, (D2-115002-3) showed that for 1973 mission, a spacecraft ERP of 79 dbm is required to support the 48,000 bps data rate. For a constant pointing error criterion of $\pm$ 0.6 degree, the spacecraft configuration required a 13.7-foot antenna and a 50-watt worst-case transmitter.

The results of this study can be used to define the spacecraft configuration required to support the higher data rates anticipated for the 1977 and 1979 missions. A 50-watt (nominal) transmitter output power, a $\pm$ 0.35 degree pointing error and a 19-foot antenna, result in an ERP of + 81.5 dbm. Increasing the transmitter power to 140 watt raises the ERP to + 85.9 dbm.

In terms of data transmission capability, this worst-case ERP value at a range of 3×10^8 km is equivalent to + 88.3 dbm nominal ERP at 3.6×10^8 km. The parameters of the 1977 and 1979 communication subsystem which can be derived from the results of this study are summarized in the table below. For a fixed transmission rate of 320,000 bps, the digital system requires an ERP level of + 92.4 dbm consisting of a 19-foot antenna and a 360-watt transmitter. The equivalent information transmission (27 kHz) in an analog system requires a spacecraft ERP of + 88.3 dbm which is implemented with a 19-foot antenna and a 140-watt transmitter, a saving in primary power of 735 watts.

For a fixed spacecraft ERP level of + 88.3 dbm, the analog system with the configuration given above is capable of transferring information equivalent to a digital rate of 320,000 bps. With this level of ERP, a digital system could transmit only 124,000 bps.

	Fixed Transmission Rate		Fixed Spacecraft ERP
Modulation Technique / Telecommunication Parameters	Digital (16,5) coded PSK/PM	Analog VSB-AM/PM or FM/PM	Digital (16,5) Coded PSK/PM
Transmitter power (watts)	360	140	140
D.C. Power	1200	465	465
Antenna Diameter (ft)	19	19	19
x Effective Radiated Power (dbm)	92.4	88.3	88.3
Data Bandwidth (kHz)	64*	27	10.3*
Data Rate (bps)	320,000	320,000*	124,000

*Equivalent parameters.

x These nominal values are derived by adding 2.4 db to the worst-case ERP scale values of Figure 3.9-4.

Comparison of all the modulation techniques considered in this study (Figure 3.9-4) indicates a 9-db (factor of 8) spread in the bandwidth capability between the digital uncoded PSK/PM technique and the analog FM carrier modulation technique. In view of the preceding considerations, a 4 to 5 db (or factor of 3) improvement over the (16,5) coded digital PSK/PM technique could be achieved with an analog modulation technique, such as VSB-AM/PM, for the 1977-79 Voyager mission profiles.

3.10 CONCLUSIONS

Significant conclusions from the mission and system analyses are as follows:

1) Based on a detailed space mechanics analysis, the 1973 mission orbit and interplanetary trajectory parameters (periapsis, apoapsis, orbit period, orbit inclination, arrival date, illumination angle, etc.) presented as baseline requirements in Section 2.0 are considered practical and achievable.

2) Should program events occur that prevent a launch within the performance envelope (launch date span), several program alternates are available, permitting a less than optimum mission. These alternate missions will accumulate useful data prior to the subsequent (2 years in the future) full-capability mission opportunity.

3) A significant increased weight (approximately 3000 pounds per planetary vehicle) in Mars orbit can be realized over the 1973 baseline mission by careful selection of launch periods, launch dates, and type of interplanetary trajectory.

4) To maximize telecommunication capability (minimum Earth-Mars distance), early Mars arrival dates should be selected and full propulsion subsystem capability utilized for the 1973 mission. An increase of 2 times in the potential data bit rate is possible within the 1973 mission design envelope by selecting the most advantageous launch in preference to an early April 1974 arrival.

5) The canister could be separated prior to orbit insertion. However, to increase confidence in the maintenance of planetary quarantine, and to simplify flight capsule thermal control, it is recommended that the canister be separated subsequent to orbit insertion.

6) All candidate propulsion units (LMDE, LMDE + C-1's, Agena, and Trans-stage engines) satisfy the Voyager spacecraft system requirements, but LMDE is preferred.

7) A quantity of 1.57 x 10^{11} bits was determined to be the required amount of data necessary to satisfy the 1973 mission scientific objectives. This amount of data for a 6-month 1973 mission orbit life results in an average data bit transmission rate of 12,500 bps.

8) A laser communication system is desirable for use during orbital mission operations to transmit larger amounts of science data to Earth. An extensive development program is believed necessary for incorporating the proposed laser system into the Voyager program. Therefore, to avoid peak expenditure of resources, the first installation should be experimental and scheduled for the 1975 mission.

9) The use of digital modulation techniques for the 1973 data link is consistent with that mission's science payload. When a large amount of analog data is generated by a single sensor (i.e., the high resolution film camera in 1977 and 1979), analog modulation techniques, such as VSB-AM/PM, provide greater transmission capability (approximately 3 times) than digital modulation techniques for a given level of effective radiated power.

4.0 VOYAGER MARS 1973 SPACECRAFT DEFINITION SUMMARY

4.0 VOYAGER MARS 1973 SPACECRAFT DEFINITION SUMMARY

The spacecraft configuration for this study was developed by using (1) the requirements and constraints presented in Section 2.0 of this document, (2) the results of the system level analysis given in Section 3.0 of this document, and (3) available data produced in previous Voyager design efforts. This developed spacecraft configuration is described in D2-115002-3. A summary of this description is included in this volume for ready reference for the operational support equipment (OSE) and mission-dependent equipment (MDE) preliminary requirements and concepts, and for the spacecraft logistic considerations discussed in Sections 5.0 and 6.0 of this volume.

The configuration was basically developed by (1) identifying the functional elements of the spacecraft, (2) determining the equipment requirements necessary to perform these functions, (3) estimating the equipment physical characteristics, and (4) integrating the equipment into an overall spacecraft configuration that satisfies the Section 2.0 requirements and constraints. The functional spacecraft elements must provide electrical power, guidance and control, data storage, telecommunications (telemetry, radio, and antenna), computing and sequencing, data storage, propulsion, structural and mechanical support, pyrotechnics, temperature control, cabling and packaging, and provision for spacecraft science. The development of the spacecraft configuration from these functional elements considered the design objectives of:

1) High packaging density.

2) Modularization of major spacecraft elements.

3) Efficient structural concepts.

4) Unobstructed viewing by science and engineering equipment (sensors and antennas).

5) Subsequent mission growth capability with minimum modifications.

Using these design objectives, the spacecraft was divided into three basic modules. These are the basic support, equipment, and propulsion modules, as shown in Figure 4-1. The basic support module is composed of the support truss and the primary spacecraft support ring. This support ring supports the guidance and control bay and other equipment, including the deployable science scan platform, and the low gain, medium gain, and high gain antennas. The equipment bays and cable tray assembly form the equipment module which attaches to the support ring on the basic support module. The removable propulsion module is the third module. This module contains and supports the capsule support structure, the main propellant tanks, the propellant pressurant tanks, the start tanks, the engine, and all associated plumbing and wiring. This module fits inside the basic support module support ring and joins that assembly at the eight apexes of the support truss. The equipment support module is stabilized by joining the upper cable tray to the upper portion of the propulsion module. The guidance and control system is installed and aligned with the spacecraft. The attitude control nozzles are mounted on the external support truss and the four nitrogen tanks are installed in pairs in the equipment module. The solar array connects between the outer circumference of the support truss of the basic support module and the inside of the propulsion module.

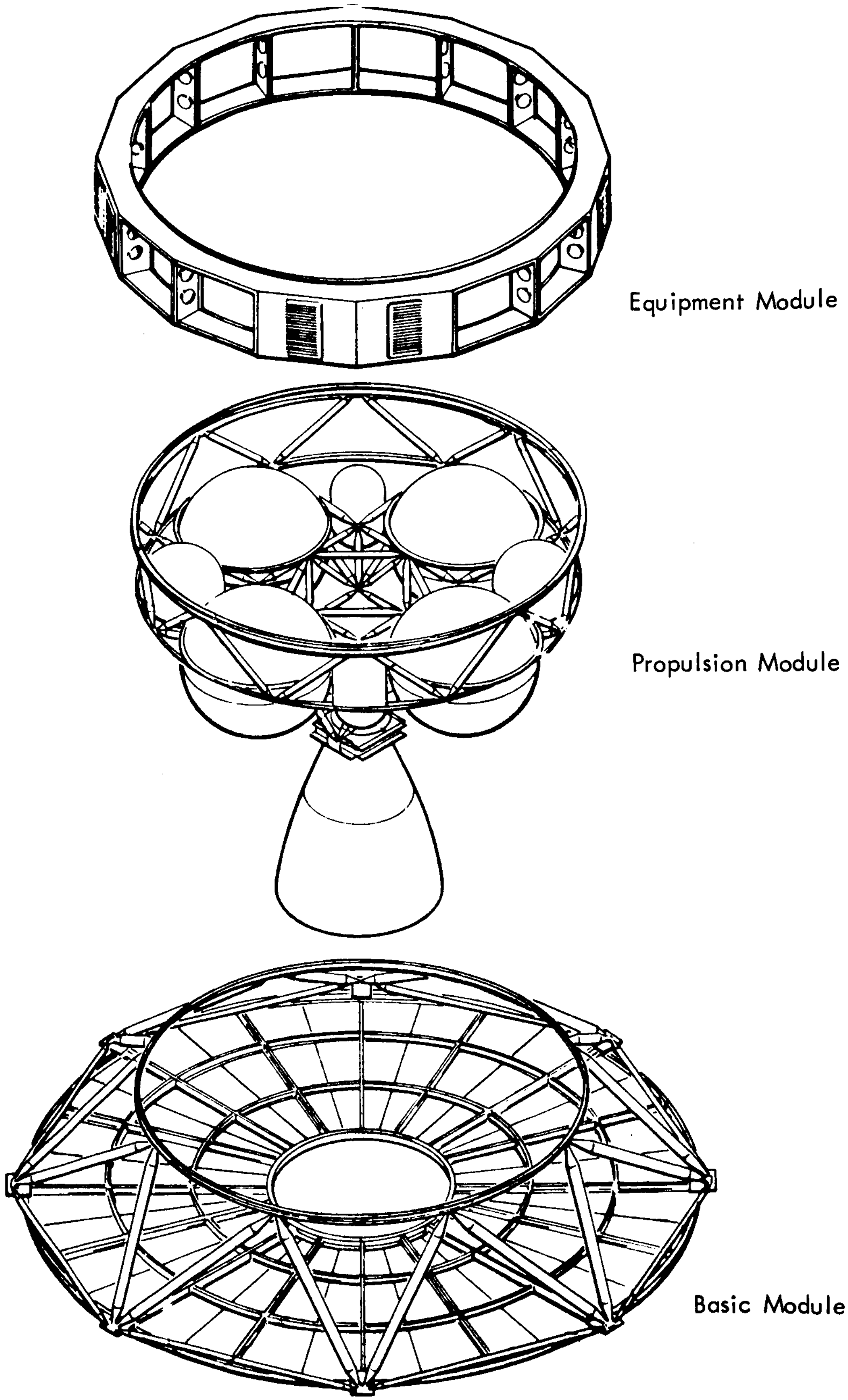


Figure 4 - 1 : BASELINE MODULES

Significant configuration features of the spacecraft are an overall diameter of 240 inches and a maximum length of 170 inches in the stowed condition. The solar array area is 290 square feet, and the high gain antenna diameter is 13.7 feet. The propulsion module engine is a modified lunar module descent engine (LMDE) using Earth-storable bipropellants. The spacecraft with the 1973 mission propellant load weighs 16,520 pounds, including a weight of 371 pounds for science instruments and DAE. Figure 4-2 is a general view drawing of the complete spacecraft. This drawing shows the stowed and deployed conditions and gives the overall dimensions, equipment arrangement, and view (cone and clock) angles for the viewing equipment.

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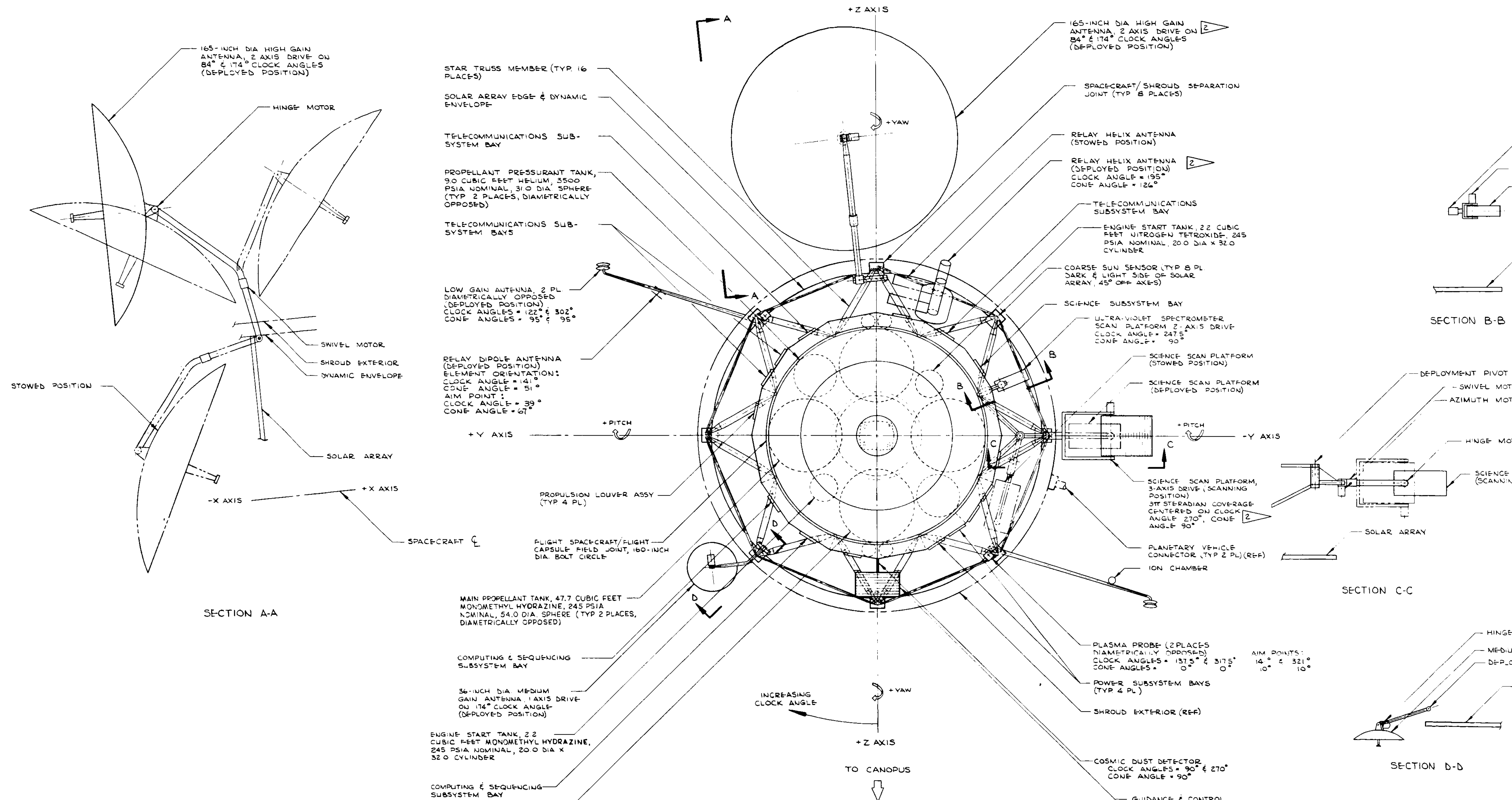


PLAN VIEW
SPACE CONFIGURATION
(LOOKING TOWARDS THE SUN)

FOLDOUT FRAME 1

FOLDOUT FRAME 2

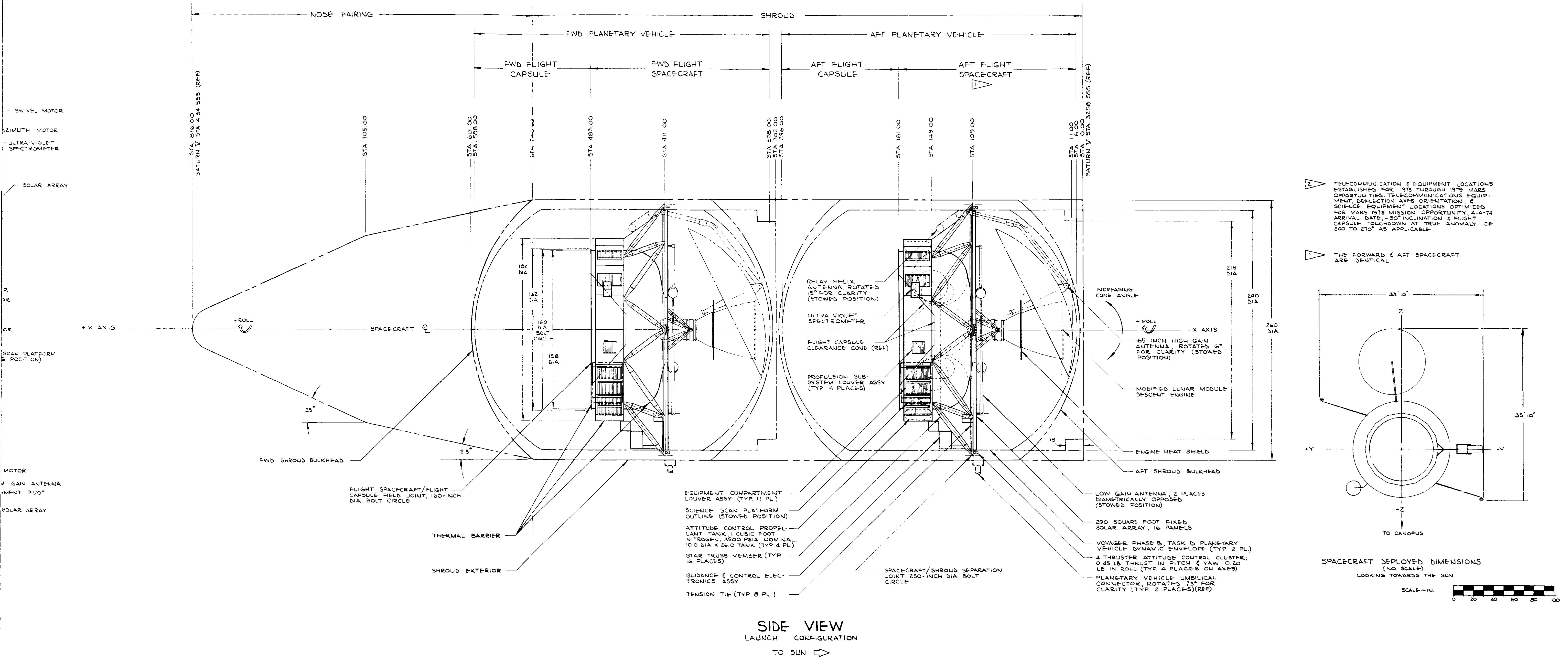


Figure 4-2: VOYAGER FLIGHT SPACECRAFT PHASE B, TASK D

FOLDOUT FRAME 3

FOLDOUT FRAME 4

FOLDOUT FRAME 5

5.0 PRELIMINARY OPERATIONAL SUPPORT EQUIPMENT AND MISSION-DEPENDENT EQUIPMENT REQUIREMENTS AND CONCEPTS

5.0 PRELIMINARY OPERATIONAL SUPPORT EQUIPMENT AND MISSION-DEPENDENT EQUIPMENT REQUIREMENTS AND CONCEPTS

Operational support equipment (OSE) must be developed that will (1) provide a high probability of success that the launch will occur within the launch window, (2) provide a high degree of confidence in the technical integrity of the spacecraft system, and (3) be economically sound. The development of OSE is affected by the geographic location for performing the various assembly and test operations and by the constraints of the flight hardware design. The development is further affected by the need to interface with equipment provided by several NASA centers, each having one or more contractors. Under these conditions, identification of OSE requirements must be accomplished by system engineering practices that account for the interface, location, and schedule constraints. The objectives of this task, i.e., to establish preliminary OSE and MDE requirements for the 1973 Voyager spacecraft, were approached with these criteria in mind.

5.1 APPROACH

For the purpose of this study, OSE was defined as that hardware and software required to support activities at Kennedy Space Center (KSC).

In developing these requirements, the following steps were taken:

1) An OSE concept compatible with the spacecraft configuration was developed.

2) Test and operation flows for the 1973 Voyager spacecraft were developed.

3) Preliminary OSE requirements and related tests and locations were identified.

4) Interfaces were identified and tabulated.

5) Commonality between OSE and MDE was determined.

6) Applicability of Apollo ACE and Saturn V ESE concepts was evaluated.

To implement these steps, it was necessary to analyze the complete system-level sequence of assembly and test operations including activities at the contractors facility prior to delivery at KSC. This approach led to the identification of factory support equipment (FSE) as well as OSE and ensured that testing operations in the factory complemented field test operations and equipment usage. Equipment requirements common to different areas of assembly and test were identified.

5.2 TEST AND OPERATIONS ANALYSIS

This analysis was directed to assembly, checkout and test, handling, transportation, capsule integration, payload module integration, and final payload test and launch operations. The flow diagrams and analysis were implemented by the approach defined in NHB 7500.1 Apollo Logistics Requirements Plan. These flow diagrams are shown in Appendix A. The test and operations analysis was performed as depicted in Figure 5-1 below:

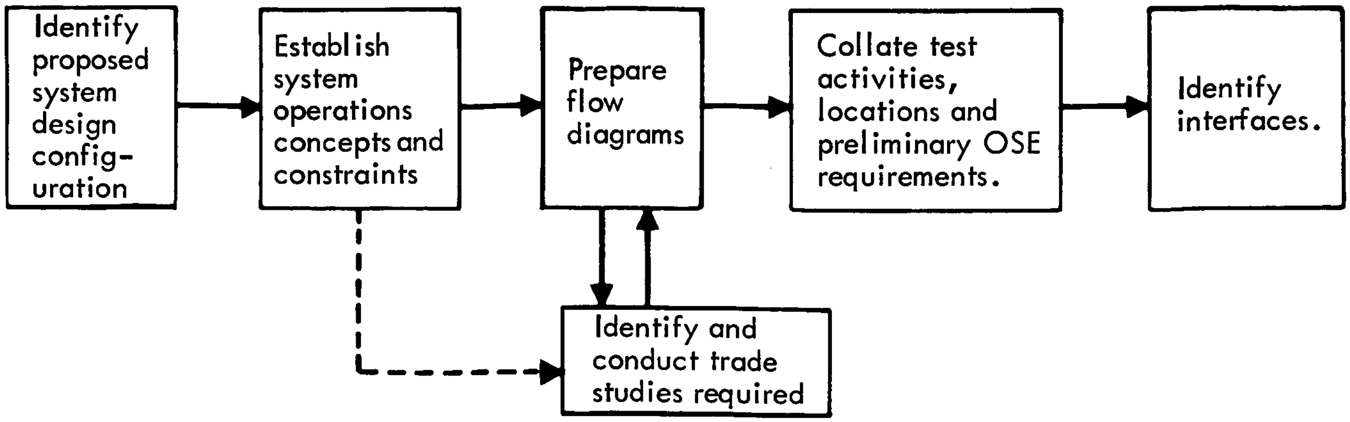


Figure 5-1: TEST AND OPERATIONS ANALYSIS

Terminology used in discussing elements of the Voyager Payload is as shown in Figure 5-2. As a prerequisite to establishing the functional flow, the operational concepts and constraints were defined. These are given in Section 5.4.1. They establish the assembly and test, transportation, handling, and prelaunch operations criteria on which the flow is based. An analysis of flight mission operations, in which MDE requirements are defined, is presented in D2-115002-4, Section 2.0.

The necessity for trade studies was revealed during the functional analysis. As an example, the operational concept initially assumed the spacecraft would be fueled at an explosive-safe area. A trade between fueling the spacecraft on the launch pad and fueling in an ESA led to selection of fueling on the pad as the preferred approach.

Having identified the constraints and concepts and conducted the necessary trades, it was possible to develop the system-level flow diagrams. These flows detailed the operations to a degree sufficient to permit preliminary definition of support hardware requirements.

OSE requirements for assembly and test operations at KSC were then identified. In addition, the performance requirements of the OSE were also established.

Review of manufacturing final assembly and test operations permitted identifi-cations of tests required to support the KSC activity. As a result, common OSE requirements were also developed. Concurrent with this analysis, preliminary equipment interfaces were identified. The preliminary OSE requirements are included in Section 5.5, and the preliminary interface identification is included in Section 5.6.

5.3 OPERATIONAL SUPPORT SYSTEM CONCEPTS

The total operational support system consists of the complex of equipment and other resources required to prepare and launch the space vehicle. For convenience, this system has been divided into four major categories:

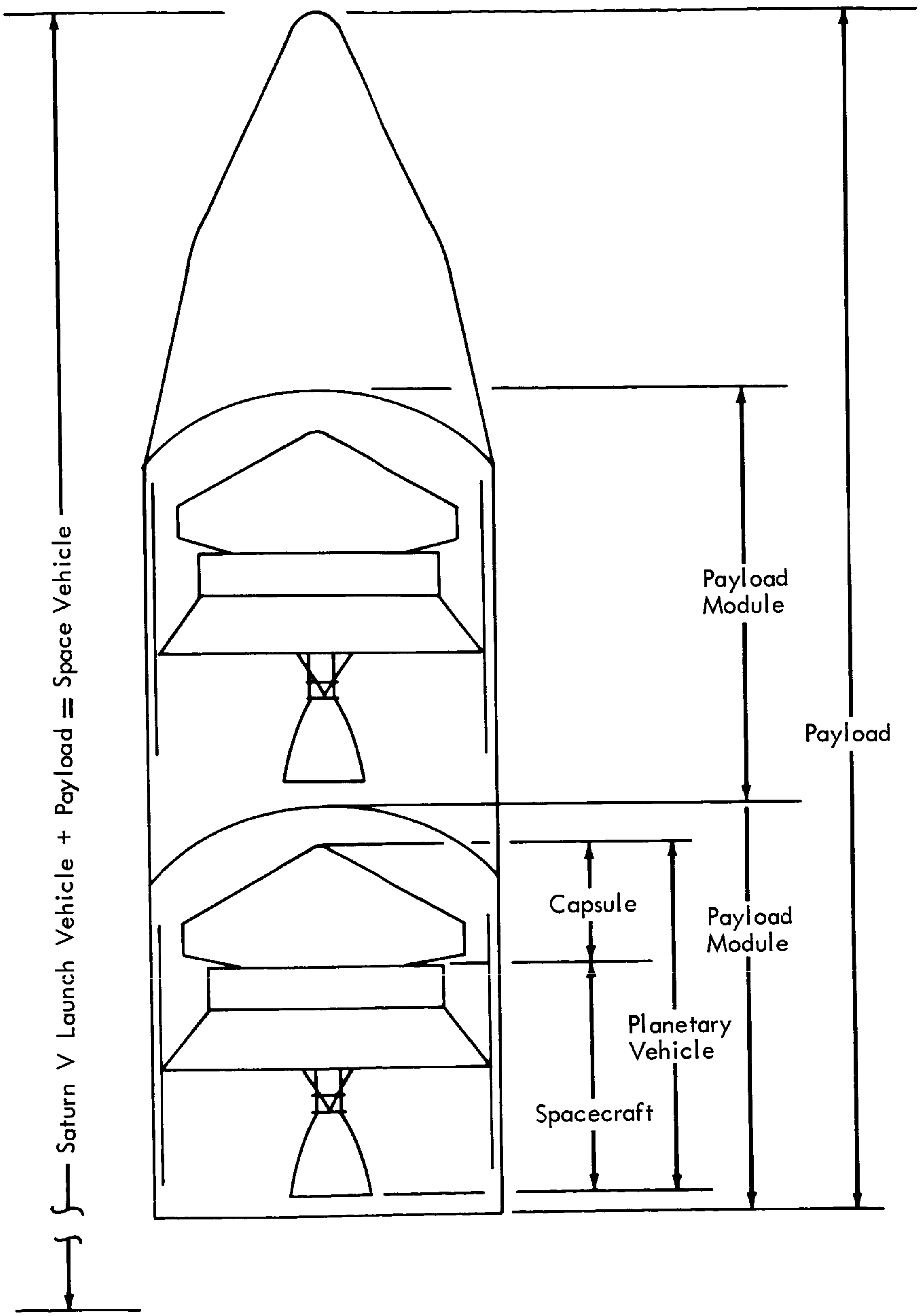


Figure 5-2: VOYAGER PAYLOAD ELEMENTS

1) The launch complex;

2) The test complex;

3) The assembly, handling, and shipping equipment;

4) The fueling and servicing complex.

An existing operational support system that is illustrated in Figure 5-3 supports current programs. The basic concept for the Voyager spacecraft OSE is to make maximum use of existing facilities and equipment and to add only that equipment required to accommodate unique features of the Voyager flight hardware. Figure 5-3 further illustrates the area in which unique Voyager spacecraft OSE must be added. Note that the Deep Space Network with its MDE is shown for continuity.

This study was to establish requirements for Voyager spacecraft unique OSE that directly supports spacecraft requirements and is compatible with the existing system. The requirements must consider the interface with other Voyager system OSE as shown in Figure 5-3.

5.3.1 Launch Control Equipment

The launch control equipment required for the Voyager spacecraft launch consists of:

1) Display and control consoles located in the launch control center.

2) Automatic monitoring and control equipment located on the launch umbilical tower.

3) Spacecraft ground power and emergency power equipment located on the launch umbilical tower.

4) S-band repeater antennas and frequency converters.

The display and control consoles enable the Voyager launch personnel to monitor status of both spacecraft and to exercise all basic control required during countdown. The major data source for the display console is the test system located in the spacecraft checkout facility.

The automatic monitoring equipment provides a continuous monitor of all critical planetary vehicle functions and automatically "safes" the planetary vehicle in case of critical malfunctions or complete loss of ground power. In addition, this equipment receives and reacts to launch vehicle initiated holds.

The spacecraft ground power system provides d.c. power to the planetary vehicle during the launch countdown. Included in the ground power system is the capability of charging the spacecraft batteries.

The S-band antennas provide the radio frequency path between the planetary vehicles and the DSIF-71. The frequency converters make it possible to use hardlines as the checkout patch between the planetary vehicles and their test systems during RF silence.

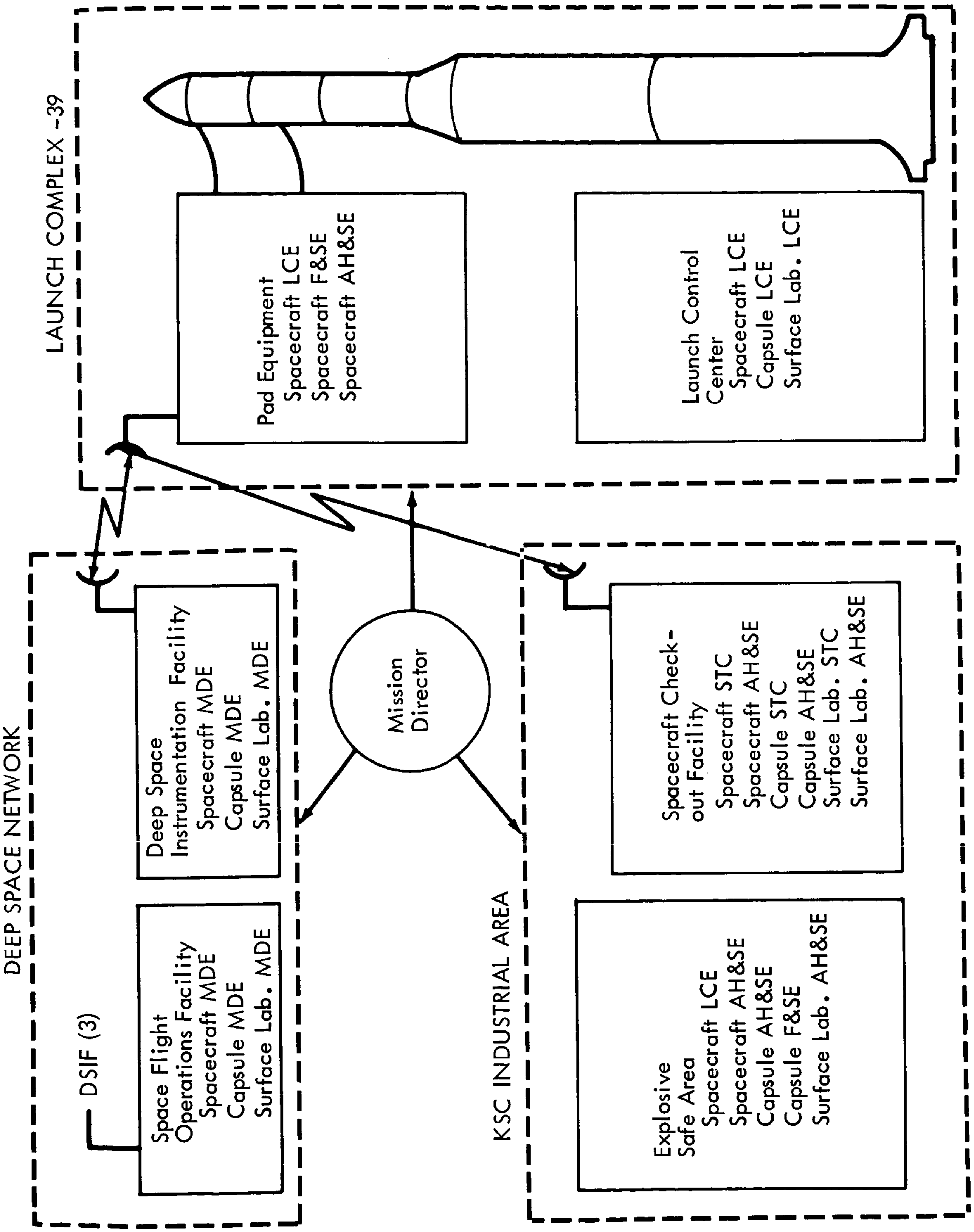


Figure 5-3: OPERATIONAL SUPPORT SYSTEM

5.3.2 System Test Complex (STC)

The STC provides the test capability required for spacecraft testing separately and when integrated to form a planetary vehicle, payload module, and payload. The STC consists of a test controller and processor, simulators, stimulators, a gimbaled two-axis test stand, and ancillary equipment items. This equipment employs a high speed general purpose computer as the test controller and data processor. The basic arrangement of this equipment and their interfaces is shown in Figure 5-4.

The test controller and processor is a computer employing a 24-bit word length and a 16-k core memory. Add-times of approximately 4 microseconds and a memory cycle time of 2 microseconds will be required.

At least four buffered input-output data channels and off-line-memory in the form of disk or drum bulk storage will be required. Auxiliary data equipment will be needed to satisfy all test requirements.

Additional signal processing equipment consists of modulators, demodulators, decommutators, decoders, and bit synchronizers. This equipment is to be a duplicate of the mission-dependent equipment.

The STC also includes power equipment that supplies ground power to the spacecraft during testing. These power supplies simulate the characteristics of the spacecraft solor panels and batteries.

Communications equipment is provided for use by the test and operations personnel to facilitate the exchange of planned and unplanned event information with remote stations as required. The simulation and stimulation equipment is used to provide signals to the spacecraft and to simulate spacecraft interfaces.

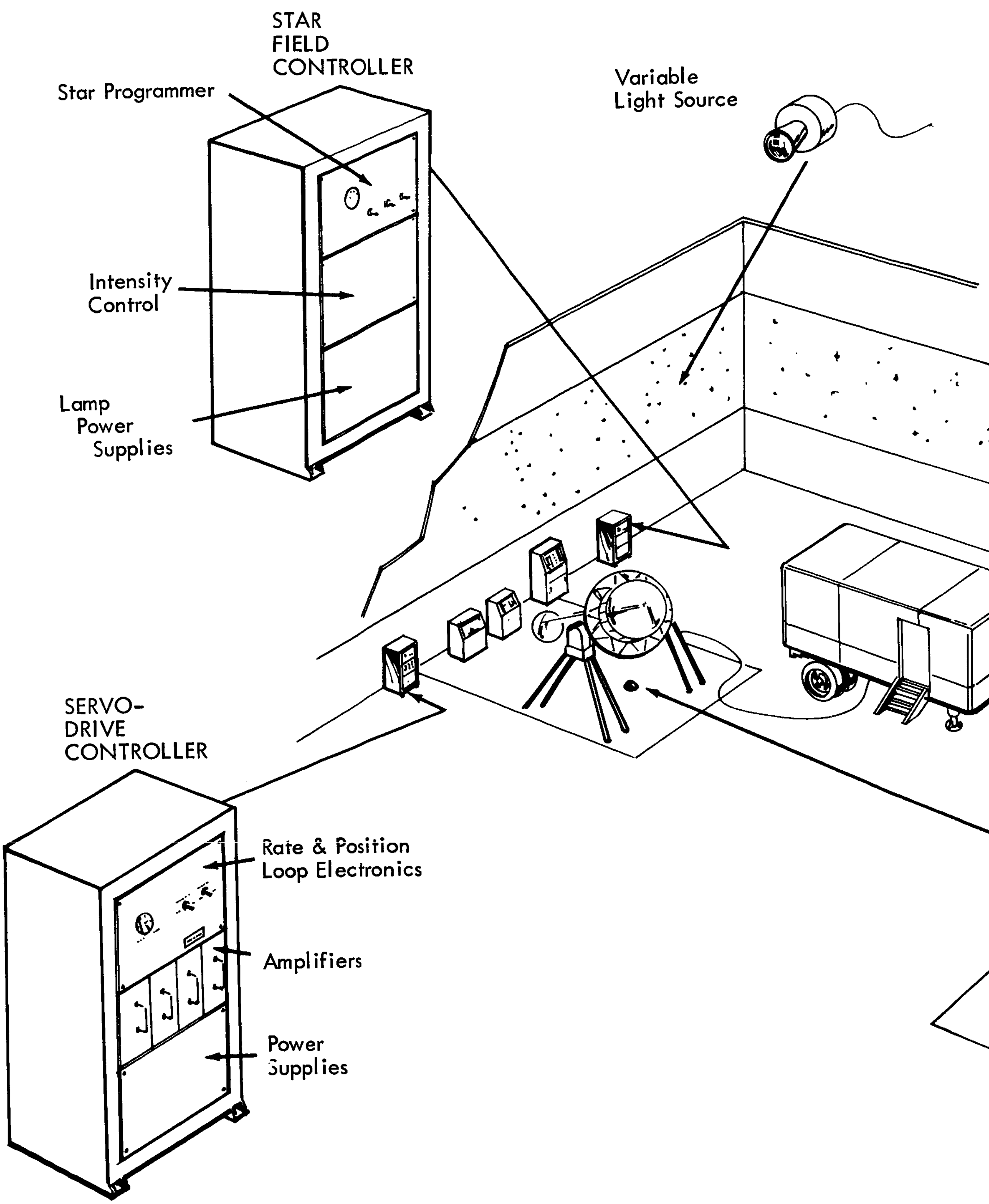
5.3.3 Assembly, Handling, and Shipping Equipment (AHSE)

The AHSE provides the capability for lifting, holding, positioning, aligning, and shipping the spacecraft, planetary vehicle, payload modules and nose fairing. Requirements to perform these functions will exist at a manufacturing assembly area during assembly and test operations and at KSC during final assembly and checkout.

5.3.4 Fueling and Servicing Equipment (F&SE)

Spacecraft fueling and servicing will include all activities required to load main propellant, attitude control propellant, and propulsion pressurants. Fuels and pressurants will be transferred to the spacecraft under Class 100,000 clean room conditions. Particulate contamination will be controlled in the propellants and pressurants. The pressurants will be loaded aseptically.

Existing facilities for fueling the Apollo service and command module hypergolic propellants will be used. These facilities are currently located on mobile service structure platforms III and IV at the 272- and 279-foot levels, respectively. Modifications will be required to extend lines for MMH and N_2O_4 to



FOLDOUT FRAME 1

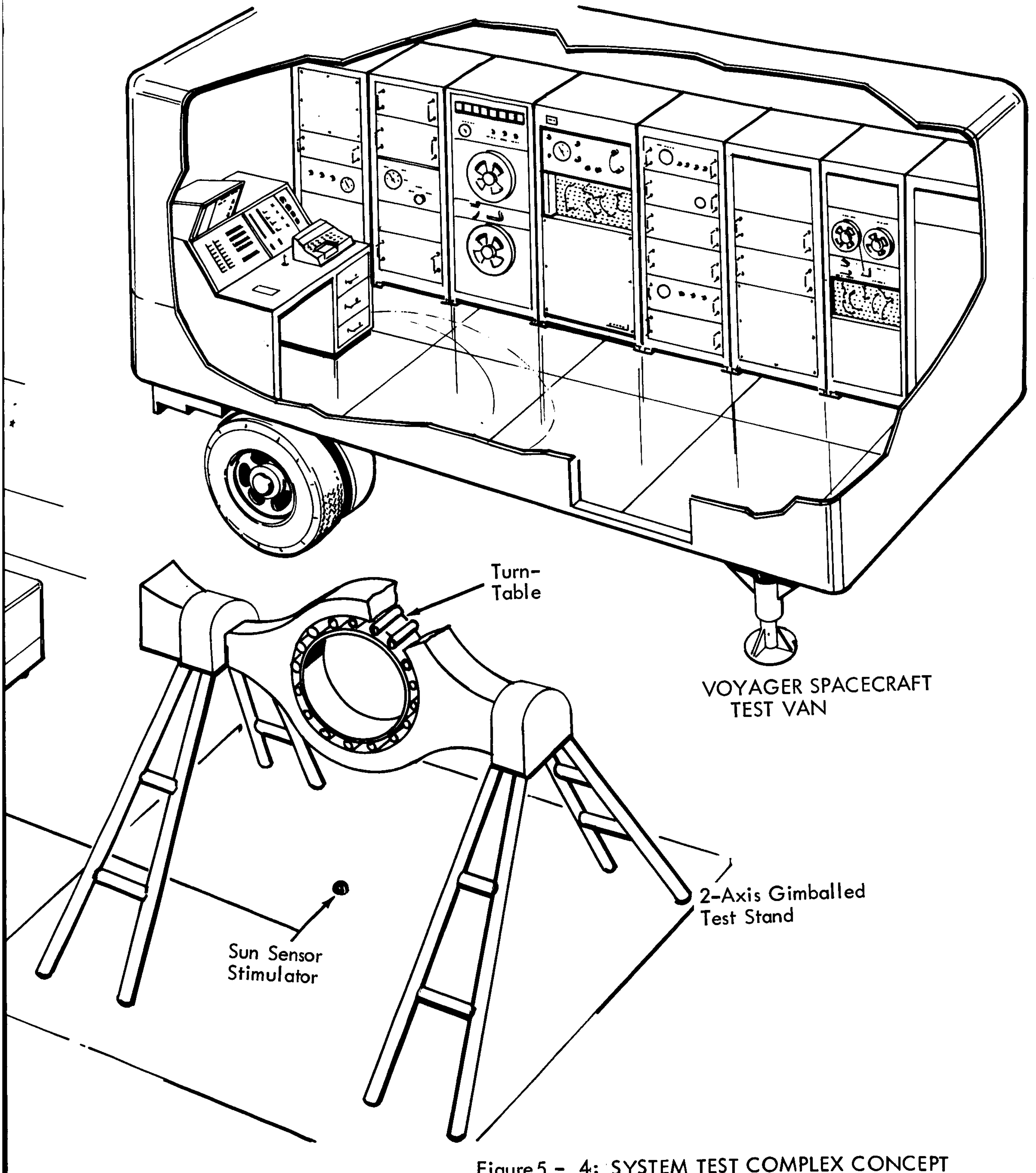


Figure 5 - 4: SYSTEM TEST COMPLEX CONCEPT

the 280- and 305-foot levels for connection to the spacecraft access locations. Facility modification will include sufficient capacity for MMH and N_2O_4 propellants for all launch opportunities. Platform access must be at these levels which will require relocation of platforms III and IV and modification for adaptation to the payload module 260-inch diameter. Facilities will be added to load the pressurants aseptically. Provisions for clean room control will be incorporated at each work platform. Existing procedure and equipment will be used for metering the quantity of fuel and oxidizer tanked. Tank calibration techniques will be used to achieve desired fuel quantity measurement accuracy.

5.4 TEST AND OPERATION FLOW DIAGRAMS

The concepts and constraints governing the activities to be defined by the operational flows were established as follows:

1) Manufacturing Assembly Operations

   a) The spacecraft structure is received at a Voyager assembly facility (VAF) in the launch attitude and resting on its own transportation support. This facility is remote from the Kennedy Space Center (KSC).

   b) The spacecraft structure, on which final assembly is begun, is an assembly consisting of the star truss, primary spacecraft support ring, and equipment bay structure.

   c) Internal insulation blankets will be installed as part of the spacecraft structure subassembly.

   d) Electronic equipment referenced in the flow includes all electronic bay subassemblies.

   e) Cabling will be delivered to the VAF as a subsystem.

   f) The science equipment will be delivered as a subsystem.

   g) The propulsion module will be delivered to the VAF ready for final assembly and installed as a unit.

   h) Storage will be provided at the VAF for components delivered prior to assembly.

   i) Solar panel simulators will be installed during manufacturing tests. Flight solar panels will be installed for acceptance test and delivery.

   j) Test batteries will be used during assembly and test.

   k) Electroexplosive device simulators will be used during assembly and test.

2) Spacecraft Acceptance Testing

   a) Acceptance tests will be conducted at the VAF.

b) The flight spacecraft and ground test spacecraft acceptance tests will include a thermal vacuum test.

c) The thermal vacuum test chamber at The Boeing Company, Space Division facility, will be used for acceptance testing.

d) The high gain antenna will be removed for thermal vacuum testing.

3) Payload Module (PLM) Assembly and Test

a) A flight capsule will be delivered to the VAF unfueled for assembly to the spacecraft.

b) The unfueled flight capsule will be assembled to the spacecraft for a joining test.

c) A planetary vehicle integrated system test will be conducted.

d) Flight shrouds will be delivered to the VAF for payload module assembly.

e) The shroud will be a structural member capable of supporting the planetary vehicle for purposes of handling and transportation.

f) The payload module will be assembled at the VAF for shipment to KSC with the following restrictions:

- The capsule will not be fueled.
- Pyrotechnics will not be installed.
- Flight batteries will not be installed.
- The propulsion system will not be fueled.
- The reaction control system will not be charged.

g) A payload module integration test will be performed.

h) Payload module environmental control will be provided during shipment.

i) The payload module will be air-transported from the VAF to the KSC skid strip.

j) The payload module will be shipped in nonlaunch attitude.

4) Final Assembly and Test at KSC

a) The PLM's will be processed at KSC in a Voyager final assembly area (VFAA).

b) The following sequence will be performed at the VFAA:

- The shroud will be removed.
- The capsule will be removed and delivered to a capsule test facility for fueling, sterilization, and test.
- Spacecraft operation will be verified by test.
- The capsule will be returned to the VFAA in a sterilized, fueled, sealed, and tested condition for integration with the spacecraft.
- The planetary vehicle will be reassembled and tested. The flight shrouds will be reinstalled and the payload module tested.
- Flight batteries and pyrotechnics will be installed during payload module buildup.
- Weight and balance tests will be conducted on the planetary vehicle and payload module using simulated fuel and oxidizer in the spacecraft.

c) A stacked payload test will be conducted as follows:

- The nose cone and spacer will be provided to assemble a payload configuration.
- A launch vehicle simulator will be used for instrument unit (IU) interfaces.

d) Following payload stack tests the payload modules, spacer and nose cone will be destacked and stored in the VFAA.

e) The payload will be stacked on the launch vehicle (L/V) at the pad and will not be processed through the Vehicle Assembly Building (VAB).

5) Launch Pad Operations

a) The payload modules, spacer, and nose cone will be transported to the launch pad in the launch attitude.

b) Individual payload module environmental conditioners will be provided during transportation to the pad and during stacking.

c) External weather protection to the payload will be provided while stacked at the pad.

d) Umbilical connections to the payload modules will be provided for cooling, purging, power, and limited signal access. There will be no RF windows in the shrouds. Telemetry data will be transmitted through the IU.

e) The spacecraft will be fueled at the pad.

f) Access will be provided through the shroud for fueling connections from the mobile service structure (MSS).

Based on the above concepts and constraints, flow diagrams were prepared and are presented in Appendix A.

The summary flow diagram (Sheet 1) categorizes the numbered functional steps into eight general phases as follows:

Phase	Functional Step Series
Assembly and test of spacecraft	11000 and 12000
Assembly and test of planetary vehicle and payload module	13000
Transport payload module to KSC	14000
Disassemble payload module and test spacecraft at KSC	15000
Assemble flight planetary vehicle and test	16000
Assemble flight payload module and test	17000
Stack flight payload test, destack, and standby storage	18000
Transport to pad, stack to space vehicle test, fuel spacecraft, and launch countdown	19000

5.5 PRELIMINARY OSE REQUIREMENTS

This section is divided into five subsections. The first subsection summarizes the general requirements applying to all OSE. The remaining four subsections itemize the specific requirements for:

- Launch control equipment
- System test complex equipment
- Assembly handling and shipping equipment
- Fueling and servicing equipment.

5.5.1 General Requirements

The operational support equipment shall:

1) Operate independently of launch vehicle electrical support equipment (ESE).
2) Acquire, process, distribute, and display spacecraft and test facility data for real-time and non-real-time analysis by the use of a general purpose computer system.
3) Perform automatic test sequencing with manual override, semiautomatic test sequencing, and manual test sequencing.

4) Provide test documentation by recording on magnetic tape.

5) Be flexible in design to accommodate configuration changes.

6) Be interchangeable with respect to servicing different spacecraft of the same design.

7) Be designed for expeditious disassembly, transport, and assembly in support of any testing required at remote test facilities.

8) Provide self-test and calibration validation.

9) Perform diagnostic test routines and isolate malfunctions to the provisioned spares level.

10) Provide critical path redundancy where applicable to ensure high probability of successful launch.

11) Contain safeguards that minimize hazard to mission hardware and operating personnel.

12) Provide automatic monitoring, interlock circuitry, and safe capability.

13) Provide emergency power sources where required.

14) Accommodate a single point ground system in accordance with that used for Saturn V at Launch Complex 39.

15) Provide central voice communications.

16) Distribute time codes from a central source.

17) Synchronize its time source to Launch Complex 39 Saturn timing equipment.

5.5.2 Launch Control Equipment

The launch control equipment shall:

1) Provide monitoring of spacecraft d.c. bus voltage levels.

2) Provide monitoring of spacecraft a.c. bus voltage levels.

3) Provide means of simulating spacecraft clock start signal from launch control equipment.

4) Provide external power to spacecraft.

5) Automatically evaluate safety-critical umbilical functions.

6) Provide emergency backup power.

7) Display spacecraft status at the launch control center.

8) Route capsule data to capsule launch control equipment.

9) Route status information to the launch director.

10) Sequence control by T-count rather than real time as required.

11) Provide automatic safeing of spacecraft in the event of complete power failure.

12) Provide automatic safeing in the event of loss of any critical control.

13) Provide computer and sequencer memory load command.

5.5.3 System Test Complex Requirements

The following are specific test parameters of the Voyager spacecraft for which STC hardware and software must be provided:

1) Evaluate d.c. bus voltage levels and noise content as a function of simulated solar panel inputs and changing loads due to spacecraft operating modes. Simulated solar panel inputs varied as a function of mission operations.

2) Evaluate a.c. bus voltage levels, frequency, and waveform as a function of simulated solar panel inputs and spacecraft operating modes.

3) Provide evaluation of the following guidance and control (G&C) parameters as a function of mission operations:

   - Sun acquisition signals
   - Sun sensor error signals
   - Star mapping signal
   - Canopus sensor error signal
   - Canopus recognition signal
   - Gyro signals.

4) Provide means to simulate input commands from computing and sequencing subsystem, 6-volt, 4.8-kHz (square wave).

5) Provide means to simulate propulsion thrust vector control feedback signals.

6) Provide means to simulate electroexplosive device fire signals from pyrotechnic subsystem.

7) Provide means to stimulate coarse and fine Sun sensors.

8) Provide means to stimulate Canopus sensors.

9) Provide means to torque gyros.

10) Provide a ground test transmitter similar to DSIF type.

11) Provide a ground test receiver similar to DSIF type.

12) Provide a means for verifying the command loop.

13) Provide capability to modulate test transmitter from prerecorded tape and/or manual command word generator.

14) Provide means of determining RF characteristics input to spacecraft radio system.

15) Provide means of simulating capsule RF inputs.

16) Provide ability to demodulate RF carrier and recover telemetry data.

17) Provide ability to determine RF characteristics generated by the spacecraft.

18) Provide means of verifying ranging operations.

19) Provide means of decoding and decommutating telemetry data.

20) Provide means of calibrating and converting the spacecraft telemetry data to engineering format.

21) Provide means of simulating the science subsystem data storage inputs.

22) Provide means of evaluating telemetry and radio outputs provided to the launch vehicle.

23) Provide means of evaluating the lower subcarrier pulse-code-modulation data stream.

24) Provide means of evaluating command detector output to computing and sequencing subsystem.

25) Provide simulated gyro rate inputs.

26) Provide means of simulating accelerometer inputs.

27) Provide means of simulating "Sun present" and "Canopus present" signals.

28) Provide means of simulating command baseband input signals.

29) Provide means of simulating antenna position signals.

30) Provide means of simulating separation signals.

31) Provide means of simulating spacecraft clock start signal from launch control equipment.

32) Provide means of evaluating computer and sequencer (C&S) outputs to other spacecraft subsystems.

33) Provide test electrical explosive devices to enable actuation of release and deployment mechanisms via pyrotechnics subsystem.

34) Provide means of determining mechanical alignment of deployables in both stowed and deployed positions.

35) Provide means of compensating for Earth gravity during testing of deployables.

36) Provide means of collimating antenna electrical and mechanical axes.

37) Provide means of collating antenna axis to Sun and Canopus axis.

38) Provide means of stimulating spacecraft transducers. A minimum of three points within the dynamic range is required.

39) Provide a means of recording the telemetry digital data versus stimulus for each point on each channel. This recording shall be in both engineering language and machine language.

40) Provide a two-axis gimbaled test stand with servo drive with the following requirements:

Freedom	- 360 degrees in roll
	- 180 degrees in pitch
Rate	- 0.02 degree -0.2 deg/sec

41) Provide star field and Sun simulation.

42) Provide spacecraft position information with respect to star field.

43) Provide means to communicate (RF) with spacecraft after acquiring Sun and Canopus.

44) Provide means of variable RF transmission delay to simulate Earth-to-Mars distance.

45) Provide means to shift transmitter frequency to simulate doppler effect.

46) Provide means to vary spacecraft bus voltage.

47) Provide a means of transmitting all spacecraft commands.

48) Provide means of evaluating spacecraft performance via telemetry downlink.

49) Provide means to simulate midcourse and orbit insertion firing response.

50) Provide means to simulate acceleration signals due to midcourse and orbit insertion firing reactions.

51) Provide means of extending deployables after acquiring Sun and Canopus reference.

52) Provide means to test the science subsystem.

53) Provide computer and sequencer memory load command.

5.5.4 Assembly, Handling, and Shipping Equipment

Assembly, handling, and shipping equipment (AHSE) will:

1) Provide containers for the following:

- Ordnance storage, handling, and shipping
- Payload module environment control pack
- Shroud umbilical simulator storage, handling, and shipping
- EED storage, handling, and shipping
- Battery storage, handling, and shipping.

2) Provide hoists, adapters, and slings for the following:

- Nose cone stacking
- Payload module assembly handling and stacking
- Payload module weight and balance fixture
- Shroud section assembly
- Planetary vehicle handling
- Capsule handling
- Planetary vehicle weight and balance fixture
- Payload module attitude rotation.

3) Provide environmental protective covers for the following:

- Payload
- Payload module
- Spacecraft
- Nose cone.

4) Provide aids, jigs, and fixtures for the following:

- Nose cone alignment and positioning
- Payload module stacking, alignment, and positioning

- Weight and balance alignment
- Shroud section assembly
- Air transport loading
- Hydraulic hoist control.

5) Provide platforms for the following:

- Payload module assembly and test
- Payload module storage
- Nose cone support and storage
- Launch vehicle simulator support
- Assembly and test area payload operations.

6) Provide additional items as follows:

- Ground transport of payload modules and nose cone
- Miscellaneous handling dollies and work stands.

5.5.5 Fueling and Servicing Equipment (F&SE)

The following specific requirements must be met by the F&SE to fuel the spacecraft on the pad in the payload stacked positions:

1) Provide a means to transfer MMH and nitrogen tetroxide from a transfer storage system at the launch pad to the spacecraft propulsion system fuel storage spheres.

2) Provide a means to charge the propulsion pressurant storage with 40 pounds of helium at 3500 psi.

3) Provide a means to charge the attitude control system with 61 pounds of GN_2 at 3500 psi.

4) Provide a means to retain spacecraft environment during fueling at Class 100,000 or better clean room conditions.

5) Provide a means to control fuel and fuel line particulate contamination. Particulate contamination shall not exceed the following absolute values:

Size (microns)	Allowable Number of Particles
Below 5	No limit
5 to 25	220
26 to 100	10
Over 100	None

6) Provide a means to filter and load pressurant such that particulate contamination shall not exceed the values listed under Item 5.

5.5.6 OSE Software Requirements

The major elements of OSE software are the computer programs required for the system tests. These are as shown in the matrix below.

	Voyager Assembly Facility	Voyager Final Assembly Area	Pad
Spacecraft	X	X	
Planetary vehicle integration	X	X	
Payload module integration	X	X	
Payload integration		X	X
Flight simulation	X	X	
Launch control equipment			

Tests at the factory and at all subsequent test locations will have common elements. The basic software program is that used for spacecraft tests at the factory. This program will accommodate the use of simulators and will provide for the most detailed of all spacecraft tests. It will also be the most exacting from the test tolerance standpoint.

As the spacecraft processing sequence approaches the launch configuration by integration into the planetary vehicle, payload module, and payload configurations, the basic test program will be modified to:

1) Accommodate flight hardware replacement of simulators.

2) Accommodate additional testing imposed by integration requirements.

3) Accommodate cone of tolerance conditions.

4) Accommodate the decreased availability of test points.

As the test program approaches the final flight condition, the test software configuration approaches that of the DSN software, thus providing for test and mission data continuity.

OSE software will consist of those programs required for real-time control of the spacecraft during tests, real-time control of the test hardware during tests, online data analysis, offline data analysis, display, test control, and executive routine required to tie these programs together. Software programs must have the ability to:

- Generate commands and decommutate the telemetry data.
- Fault-isolate to the replaceable level consistent with the phase of testing involved.
- Provide for repetitive testing of a given subsystem while varying the parameter to which the particular subsystem is most sensitive.
- Provide output data in engineering terms with all out-of-tolerance flagged.
- Determine event occurrence to permit trending of test data.
- Permit either computer-controlled or operator-controlled modes of testing.
- Permit self-check of the computer and all peripheral test equipment.

The system test software will also include a flight simulation program whose basic purpose is to permit evaluation of spacecraft performance during environmental tests. The software will supply the reference against which spacecraft performance is compared. This program is intended primarily for use during environmental tests, but will be applicable to any test location. As can be noted from the above table, there are no software programs specifically associated with the launch control equipment.

5.6 INTERFACE IDENTIFICATION

Interfaces between the functional areas of the spacecraft system and the launch vehicle, capsule bus, surface laboratory, launch operations, tracking and data acquisition, and mission operations systems have been identified. At this level, these interfaces are as shown in Figure 5-5.

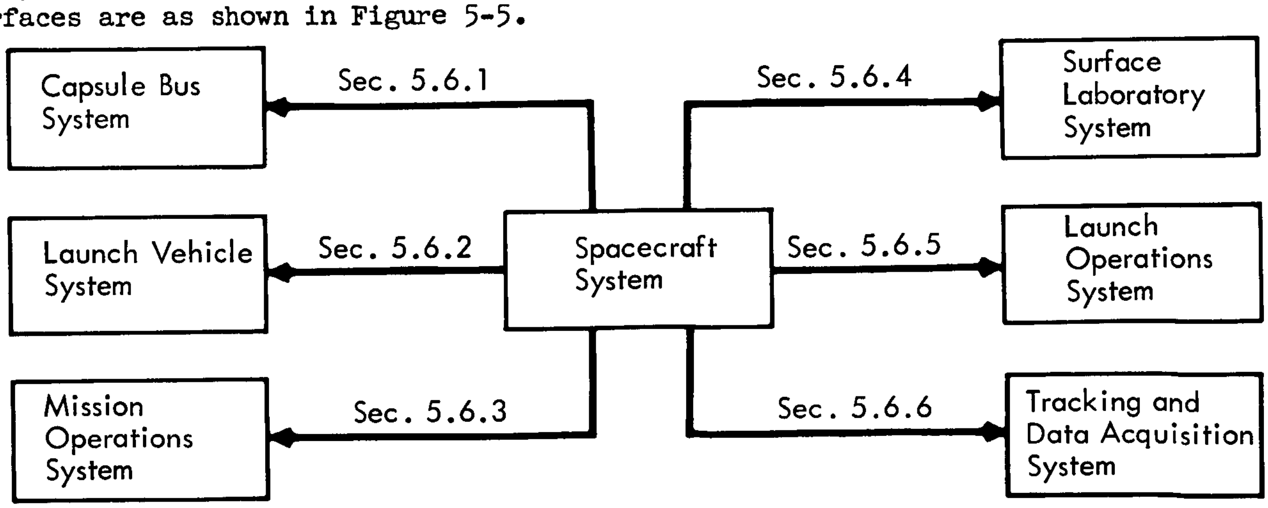


Figure 5-5: SPACECRAFT SYSTEM INTERFACES

Further detail of these interfaces is presented in Section 1.1.3 of D2-115002-3. The section numbers shown in the figure relate to subsequent paragraphs that discuss the interfaces in greater detail.

5.6.1 Spacecraft System-to-Capsule System Interface

The spacecraft and capsule systems have physical, power, signal, and thermal interfaces between the capsule and the flight spacecraft as follows:

1) The physical interface between the capsule and spacecraft consist of a structural field joint and the electrical and RF connectors on the interconnecting cables.

2) The power interface consists of four lines carrying a maximum of 200 watts. This voltage will vary from 37 to 108 volts.

3) The signal interface consists of approximately 60 lines, including spares. Eight of these are assigned to the telemetry downlink system for transmission of capsule engineering data. Approximately 33 are reserved for command and control functions. Capsule engineering data is converted to digital form in the capsule and transmitted to the spacecraft at a rate of 5.8, 9.2, or 115.5 bps. Command signals are in the form of 6-volt, 4.8-kHz square waves.

4) The thermal interface occurs at the exterior surface of the capsule. Cooling air will be supplied to the capsule exterior during ground operations.

5) Spacecraft OSE will include a capsule simulator designed to simulate all of the above interfaces.

6) The spacecraft and capsule systems also have external signal interfaces within the system test complex and in the launch control equipment.

In addition, there are four data interfaces between the spacecraft OSE and capsule OSE as follows:

1) Capsule science data derived from the upper subcarrier by demodulation and decoding in the spacecraft OSE is relayed to the capsule OSE.

2) Capsule engineering data via the lower subcarrier demodulators and format identifiers is relayed to the capsule OSE.

3) RF links to the spacecraft relay receiver antenna for simulation of capsule-to-spacecraft data transmission.

4) Capsule command data through the spacecraft OSE. (A design objective for the spacecraft OSE should be to maintain the same characteristics for the 60-line signal interface after planetary vehicle assembly as was seen prior to spacecraft-to-capsule mating).

5.6.2 Spacecraft System-to-Launch Vehicle System Interface

Physical, signal, and RF interfaces exist between the launch vehicle and the spacecraft. These are:

1) The planetary vehicles are structurally tied to the launch vehicle at eight points of attachment between the spacecraft integral adapter and the shroud.

The same attachment points are used by spacecraft assembly, handling, and shipping equipment for structural support of the spacecraft during all assembly, test, and shipping operations.

2) Signal and RF interfaces occur at the electrical and RF connectors between the lower planetary vehicle and launch vehicle instrument unit. Signal interconnections serve to transmit sensor data such as vibration, acoustic, and temperature measurements required for launch evaluation purposes. RF connections to the instrument unit provide for communication between the spacecraft and the DSIF-71 during the launch phase. Radio communication during this period will be via the instrument-unit-mounted antennas.

3) No interfaces between the spacecraft OSE and launch vehicle OSE have been identified.

5.6.3 Spacecraft System-to-Mission Operations System Interface

The mission operations system interfaces with the spacecraft through its RF system and the MDE used in support of Voyager at the DSN site.

5.6.4 Spacecraft System-to-Surface Laboratory System Interface

For the purpose of this report, all interfaces between the spacecraft and surface laboratory have been considered to be an element of the spacecraft to capsule system interface.

5.6.5 Spacecraft System-to-Launch Operations System Interface

This interface includes physical, power, signal, RF, and environmental control interfaces between the planetary vehicle and the launch operation system at the umbilical connections and again at the junction between these lines and the spacecraft launch control equipment.

5.6.6 Spacecraft System/Tracking and Data Acquisition System (TDAS) Interface

This interface includes the RF link between the spacecraft and the tracking and data acquisition system. It also includes power, signal, and physical interfaces with the spacecraft MDE.

5.7 OSE/MDE COMMONALITY

The test system and the MDE have several functional requirements in common. It is desirable that they utilize identical equipment to meet these requirements. Tabulated below are the hardware elements of MDE. These are discussed in detail in Volume IV, Section 2.0 of this report.

Item	Use in STC
Upper Subcarrier Demod (and Filter)	X
Upper Subcarrier Synchronizer	X
A/D Converter	X

Item	Use in STC
Block Decoder	X
Recorder	X
Filter (two-Channel Demod)	X
Lower Subcarrier Demod (and Filter)	X
Lower Subcarrier Synchronizer	X
Buffer and Formatter	X
Tape Assembly	
Test Selector	
Block Comparator	
Test Patch Panel	
Data Printer	
Control Panel	
Real-Time Alarm	
Computer and Sequence Display Console	

These elements of MDE indicated above have a direct impact on signal demodulation, decoding, and formatting and fulfill identical requirements in both MDE and OSE. They should be used in both applications. MDE elements not considered common are those unique to the local operation of the MDE but that have no direct impact on signal processing.

The use of common hardware elements in OSE and MDE is of direct benefit in correlating data. However, the broader aspects of commonality, or at least similarity, between mission operations and ground test operations should be considered. In the latter case, similarity of software, data handling techniques, and recording techniques is worthy of serious consideration. Any steps taken to simplify correlation of data from all sources will contribute to a more effective mission operation.

5.8 APOLLO ACE AND SATURN ESE CONCEPT EVALUATION

The Apollo ACE and Saturn ESE concepts were evaluated for applicability to the 1973 Voyager program. Basically, both concepts are the same, i.e., automated testing through the use of general purpose computers to process data between operator control consoles and special purpose equipment that interfaces with the item under test. In both cases, the functions of the special purpose equipment are the same, i.e., converting test item data to a form required by the item under test. Both systems are fixed installations that can be a handicap if testing is required at several locations. They employ the serial, preprogrammed test approach in which tests are run on a step-by-step basis and the evaluation is made by comparing test results with preprogrammed limits. They also provide the capability of parallel subroutine testing by time-sharing the computer central processors.

The Voyager spacecraft OSE should use the same basic concept as the Apollo-ACE and the Saturn-ESE, i.e., automated testing involving operator consoles, computers, and special purpose interfacing equipment.

Preliminary studies for a hardware design that would meet the OSE preliminary requirements previously discussed indicate that several operational factors must be seriously considered. These are:

1) A program schedule that calls for single launch at 2-year intervals.

2) The number of different locations at which tests must be conducted and the desirability of using the same equipment at all locations.

3) The complexity of the testing required.

4) The impact on facilities from the standpoint of modification required and availability for an extended but intermittent program.

5) Total costs of new design, modifications, relocations, software, and operations.

Considering these factors, a test system based on the Apollo ACE concept can be implemented. Such a system, using today's state of the art, could be small enough to be made mobile and thus reduce schedule, location, and facilities impacts.

The Apollo ACE and Saturn ESE concepts could be applied to the Voyager program. The application of the hardware would impose several program limitations. Among the limitations are:

1) Schedule restraints

2) Operational inflexibility

3) Difficulty in data correlation

4) Long-term facilities impacts.

The OSE concept applied to the Voyager program will avoid these limitations by employing mobile test system approach within the framework of the basic Apollo/Saturn test concepts.

6.0 SPACECRAFT LOGISTICS CONSIDERATIONS

6.0 SPACECRAFT LOGISTICS CONSIDERATIONS

6.1 OBJECTIVE

The objective of this task was to evaluate the test and operational flows for the Voyager spacecraft and to identify potential logistics problems.

6.2 APPROACH

The test and operational flows shown in Appendix A were reviewed in detail with particular attention given to the general areas of assembly and test, site support, and transportation. In general, the approach was to define problems that could not readily be solved by normal design practice and logistics planning. Such problems generally fall into one of the following categories:

1) Problems that involve an interface with other contractors or government agencies.

2) Problems that involve other programs from a schedule standpoint and hence create questions as to equipment availability.

3) Problems that are common to all elements of the total Voyager program and that should have a common solution.

6.3 PROBLEM AREAS AND RECOMMENDED SOLUTIONS

Analyses of the test and operations flows have disclosed the following logistics problems.

6.3.1 Spacecraft Fueling Procedure

Use of the Apollo lunar module fueling system for spacecraft fueling on the launch pad will require facility modification to the mobile service structure (MSS).

Discussion -- It is desirable to fuel the spacecraft at the launch pad and to utilize the existing facilities to the maximum extent possible. This approach discloses the following problem areas:

1) Mobile service structure platform levels are not in the right position for spacecraft fueling.

2) Fuel and oxidizer lines are not physically compatible with Voyager requirements.

3) Propellant storage is not sufficient for the Voyager requirement.

4) The fuel metering system may require modification.

5) Weight verification of the fueled vehicle may be required.

Recommendations -- The mobile service structure and the lunar module fueling system should be reexamined and modification requirements established. Schedule and hardware interferences with the Apollo program or with other programs must be identified so that any proposed modification is consistent with all requirements.

6.3.2 Payload Module Integration

The functional flows show that payload module integration is accomplished at the spacecraft contractor's facility. The handling and shipping of the capsule, shrouds, and payload modules impose unique transportation requirements.

Discussion

It is deemed desirable to minimize the amount of assembly and integration testing at KSC. Therefore, most of these operations will be accomplished at some other location. This requires that the spacecraft, capsule, and shrouds be shipped and handled separately and possibly as a complete payload module. An examination of this situation discloses several potential problem areas:

1) The structural features of the shroud may preclude horizontal positioning of the payload module.

2) The size of a complete payload module precludes shipment by road or rail. Structural features of the shroud may preclude shipment by air.

3) The number of assembly and disassembly operations may adversely affect payload module reliability.

4) For the total Voyager program standpoint, cost effectiveness of multiple handling and shipping operations becomes a factor.

Recommendations -- The handling and transportation requirements for the spacecraft, capsule, shrouds and payload module should be examined in detail. Particular emphasis should be directed to:

1) Modes of transportation existing and available and their adaptability.

2) Shroud structural features and possible modifications.

3) Schedule impacts of possible alternates.

4) Cost and reliability implications.

6.3.3 Payload Stack Test

A payload stack test of the flight ready payload modules and nose cone in the Voyager final assembly area will require a special facility for lifting, stacking, and tests.

Discussion -- The payload modules and nose cone require an integration test in the full payload configuration prior to installation on the launch vehicle. This test is required to minimize test time at the launch pad.

Several equipment and facilities requirements of a specialized nature are imposed by this test. Some of these are:

1) Room height must be sufficient to accommodate hoisting facilities having a hook height of approximately 85 feet.

2) A Class 100,000 clean room requirement must be met.

3) Safety requirements relating to a fueled capsule must be met.

4) A number of special fixtures and handling devices must be provided to permit safe and effective handling of the various payload elements.

Recommendations -- The specific facility, such as the pyrotechnic installation building (PIB), in which this test may be performed, must be identified, because it will impose constraints on equipment and provide interface definition. Modifications and additions required must also be identified and planned. Schedule impacts on other programs must be identified and accounted for.

6.3.4 Cleanliness and Contamination

The requirement for processing of the spacecraft as well as the assembled payload modules in a clean facility creates unique handling and transportation problems.

Background -- The logistics of handling the relatively large payload modules in a clean facility in accordance with Federal Standard 209a require specific attention. Handling equipment currently designed for use in a clean room has generally been sized for relatively small parts or subsystems. Existing equipments having the capability of handling the payload elements may contribute excessively to the contamination level in a clean facility and therefore may not be suitable for this application.

There does not appear to be a single, cost-effective solution to problem stated. The requirement must be met by a combination of:

1) Careful analyses and planning to reduce the amount of equipment required.

2) Modification of existing equipment to meet minimum requirements.

3) Development of workarounds to minimize the impact of the equipment problem.

4) New design in those areas that cannot be accommodated in any other manner.

Recommendations -- This problem should be the subject of a detailed study that will identify hardware requirements, establish procedures, and conduct the trades required to define the modifications, constraints, and new designs required.

6.3.5 Training

Training of contractor and NASA personnel will be required to ensure continued availability of qualified fabrication, maintenance, launch, and flight operations personnel. The training problem is increased by the biennial flight program.

Discussion -- Manufacturing personnel will be skilled craftsmen requiring training and certification in specialized fields. Crews operating tracking stations will require certification and training in the maintenance and operation of special equipment. The launch and flight operations require training in mission operations. The biennial mission schedule creates the necessity for retraining of experienced personnel or training of new people for each mission.

Recommendations -- Develop a comprehensive training program that will:

1) Provide qualified personnel at the launch complex on a scheduled basis.

2) Provide qualified personnel at the DSN for the conduct of each mission.

3) Provide qualified personnel at the contractors facility to ensure continuity and system growth.

6.3.6 Payload Module Malfunction

Critical launch windows require that a payload module failure on the pad is rapidly corrected. Follow-on repair activates on failed payload modules must be preplanned to ensure a backup capability.

Discussion -- The relatively short launch window available for Voyager, coupled with the long interval (approximately 2 years) between windows, makes it imperative that all possible steps be taken to ensure a successful launch. This imposes the requirement for detailed plan of payload modules replacement and subsequent repair and retest of the faulty module. This, in turn, requires a trade between onsite capability and utilization of vendor standby capability.

Recommendation -- The logistics plan for spares, utilization of vendor capability, onsite repair capability, onsite test capability, and shipping must be evaluated. This evaluation should include a detailed analysis of possible failure modes and the optimum actions to be taken.

6.3.7 Interface Between the Payload and Launch Operations System

The logistics interface between the Voyager payload and the launch operation system is chiefly concerned with the facilities on the mobile launcher and the mobile service structure and their utilization.

Discussion -- The mobile launcher and mobile service tower must be capable of accomplishing the following:

1) Lifting and stacking the payload modules, shroud, and nose cone.

2) Providing environmental protection to the payload modules.

3) Fueling and charging pressurized systems in the payload modules.

4) Providing work platforms (three levels) around the launch vehicle and payload.

5) Providing umbilical service to the payload modules.

These requirements will be met by a combination of new equipment from several sources, government-furnished equipment, and existing equipment. Operating procedures, spares procurement, shipping and handling requirements, and schedule implementation must all consider these factors.

Recommendation -- A detailed analysis of pad operations should be made. It should identify any additional procedures, equipment, and special shipping and handling problems not being accommodated in the Apollo program.

6.3.8 Spares Considerations

Sparing is a problem because of the long interval between launches. The problem is whether to purchase for all launches simultaneously or to purchase for one launch at a time.

Discussion -- A maximum initial purchase of spares gives the assurance that parts are available and are relatively independent of suppliers' changing products or going out of business. On the other hand, obsolescense, shelf-life deterioration, costs of storage, and costs of periodic test must be considered.

Recommendation -- In spite of possible cost disadvantages, spares procurement should follow the same sequence as spacecraft hardware procurement.

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APPENDIX

FLIGHT SPACECRAFT
OPERATIONS AND TEST
FUNCTIONAL FLOWS

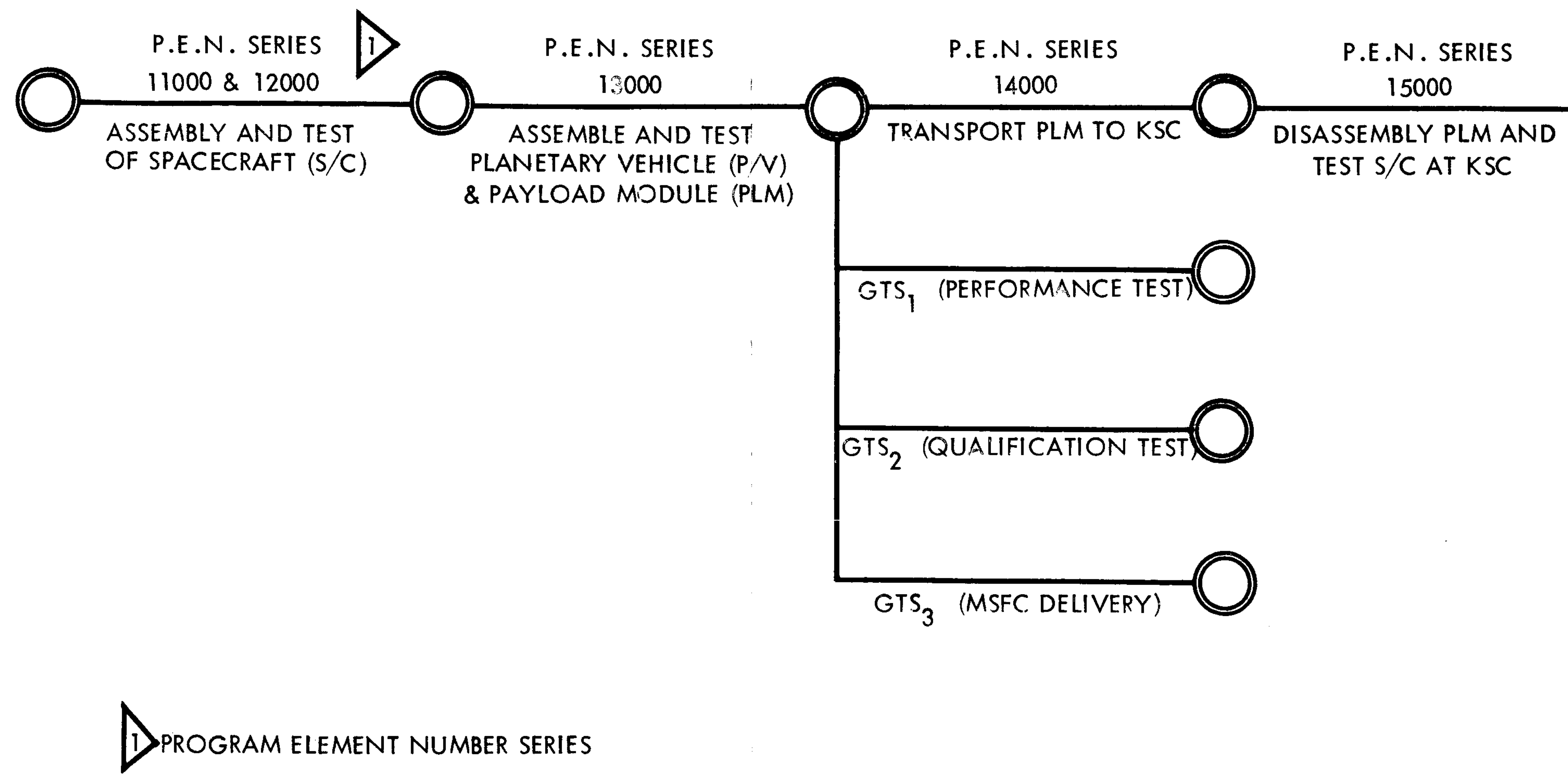
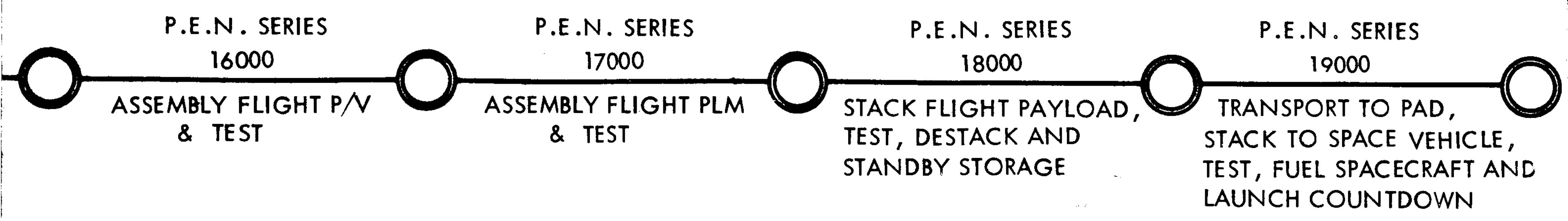

P.E.N. SERIES
11000 & 12000
ASSEMBLY AND TEST
OF SPACECRAFT (S/C)
P.E.N. SERIES
13000
ASSEMBLE AND TEST
PLANETARY VEHICLE (P/V)
& PAYLOAD MODULE (PLM)
P.E.N. SERIES
14000
TRANSPORT PLM TO KSC
P.E.N. SERIES
15000
DISASSEMBLY PLM AND
TEST S/C AT KSC
GTS1 (PERFORMANCE TEST)
GTS2 (QUALIFICATION TEST)
GTS3 (MSFC DELIVERY)
1 PROGRAM ELEMENT NUMBER SERIES

FIG A-1



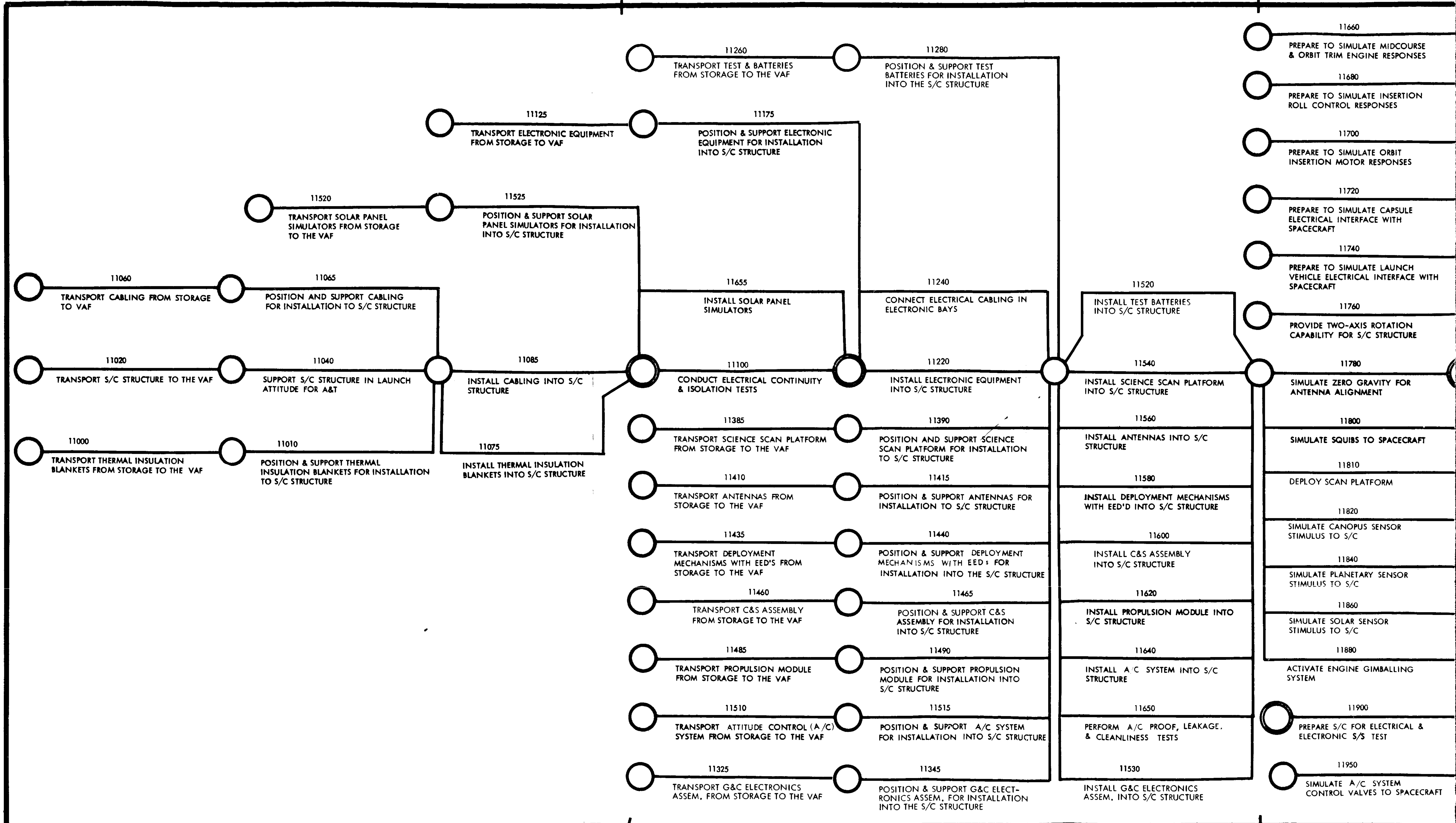
ABBREVIATIONS:

A/C - Attitude Control
A&T - Assembly & Test
CC&S - Command, Computing & Sequencing
CLT - Cargo Lift Trailer
EED's - Electrical Explosive Devices
G&C - Guidance & Control
GTS - Ground Test Spacecraft
KSC - Kennedy Space Center
LCE - Launch Complex Equipment
LOS - Launch Operations System
L/V - Launch Vehicle
PLM - Payload Module (P/V + Shroud)
L PLM - Lower PLM
U PLM - Upper PLM
P/L - Payload (U PLM + L PLM + Spacer & Nose Cone)
P/V - Planetary Vehicle (Spacecraft & Capsule)
S/C - Spacecraft
S/V - Space Vehicle (P/L + L/V)
VAF - Voyager Spacecraft Assembly Facility (Assembly Located at Kent)
VFAA - Voyager Final Assembly Area (at KSC)

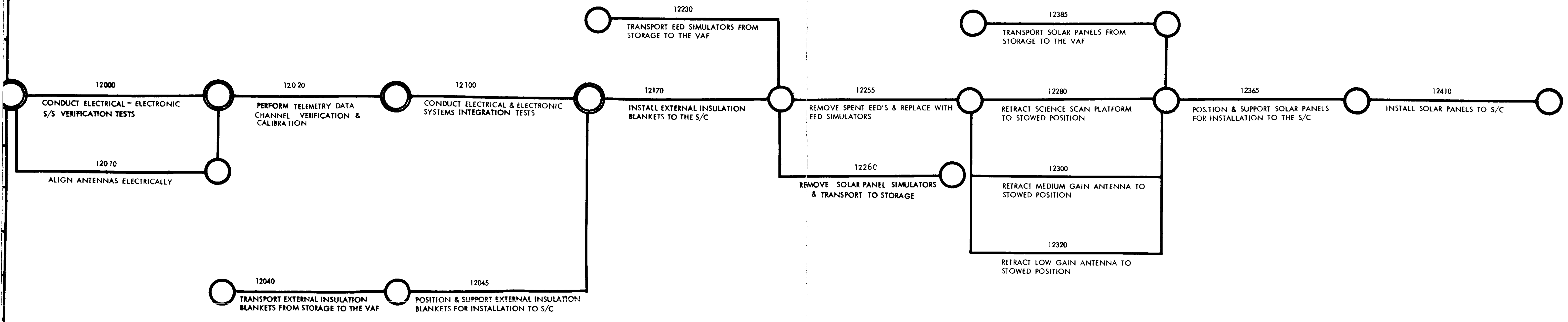
This functional flow depicts the processing of the Voyager Flight Spacecraft from Manufacturing to launch. The functional flow is made up of activities arranged on a functional requisite basis with no time sequence relationship shown.

Rev.	Date	THE BOEING COMPANY			
			Initials	Date	VOYAGER SPACECRAFT
		Drawn			SUMMARY FLOW DIAGRAM
		Checked			
		Sys. Eng.			
		Super.			
		Approved			
		Approved			

BOEING NO. FIG A -1
SH



11260
TRANSPORT TEST & BATTERIES FROM STORAGE TO THE VAF
11280
POSITION & SUPPORT TEST BATTERIES FOR INSTALLATION INTO THE S/C STRUCTURE
11125
TRANSPORT ELECTRONIC EQUIPMENT FROM STORAGE TO VAF
11175
POSITION & SUPPORT ELECTRONIC EQUIPMENT FOR INSTALLATION INTO S/C STRUCTURE
11520
TRANSPORT SOLAR PANEL SIMULATORS FROM STORAGE TO THE VAF
11525
POSITION & SUPPORT SOLAR PANEL SIMULATORS FOR INSTALLATION INTO S/C STRUCTURE
11060
TRANSPORT CABLING FROM STORAGE TO VAF
11065
POSITION AND SUPPORT CABLING FOR INSTALLATION TO S/C STRUCTURE
11655
INSTALL SOLAR PANEL SIMULATORS
11240
CONNECT ELECTRICAL CABLING IN ELECTRONIC BAYS
11520
INSTALL TEST BATTERIES INTO S/C STRUCTURE
11020
TRANSPORT S/C STRUCTURE TO THE VAF
11040
SUPPORT S/C STRUCTURE IN LAUNCH ATTITUDE FOR A&T
11085
INSTALL CABLING INTO S/C STRUCTURE
11100
CONDUCT ELECTRICAL CONTINUITY & ISOLATION TESTS
11220
INSTALL ELECTRONIC EQUIPMENT INTO S/C STRUCTURE
11540
INSTALL SCIENCE SCAN PLATFORM INTO S/C STRUCTURE
11000
TRANSPORT THERMAL INSULATION BLANKETS FROM STORAGE TO THE VAF
11010
POSITION & SUPPORT THERMAL INSULATION BLANKETS FOR INSTALLATION TO S/C STRUCTURE
11075
INSTALL THERMAL INSULATION BLANKETS INTO S/C STRUCTURE
11385
TRANSPORT SCIENCE SCAN PLATFORM FROM STORAGE TO THE VAF
11390
POSITION AND SUPPORT SCIENCE SCAN PLATFORM FOR INSTALLATION TO S/C STRUCTURE
11560
INSTALL ANTENNAS INTO S/C STRUCTURE
11410
TRANSPORT ANTENNAS FROM STORAGE TO THE VAF
11415
POSITION & SUPPORT ANTENNAS FOR INSTALLATION TO S/C STRUCTURE
11580
INSTALL DEPLOYMENT MECHANISMS WITH EED'D INTO S/C STRUCTURE
11435
TRANSPORT DEPLOYMENT MECHANISMS WITH EED'S FROM STORAGE TO THE VAF
11440
POSITION & SUPPORT DEPLOYMENT MECHANISMS WITH EEDs FOR INSTALLATION INTO THE S/C STRUCTURE
11600
INSTALL C&S ASSEMBLY INTO S/C STRUCTURE
11460
TRANSPORT C&S ASSEMBLY FROM STORAGE TO THE VAF
11465
POSITION & SUPPORT C&S ASSEMBLY FOR INSTALLATION INTO S/C STRUCTURE
11620
INSTALL PROPULSION MODULE INTO S/C STRUCTURE
11485
TRANSPORT PROPULSION MODULE FROM STORAGE TO THE VAF
11490
POSITION & SUPPORT PROPULSION MODULE FOR INSTALLATION INTO S/C STRUCTURE
11640
INSTALL A/C SYSTEM INTO S/C STRUCTURE
11510
TRANSPORT ATTITUDE CONTROL (A/C) SYSTEM FROM STORAGE TO THE VAF
11515
POSITION & SUPPORT A/C SYSTEM FOR INSTALLATION INTO S/C STRUCTURE
11650
PERFORM A/C PROOF, LEAKAGE, & CLEANLINESS TESTS
11325
TRANSPORT G&C ELECTRONICS ASSEM. FROM STORAGE TO THE VAF
11345
POSITION & SUPPORT G&C ELECTRONICS ASSEM. FOR INSTALLATION INTO THE S/C STRUCTURE
11530
INSTALL G&C ELECTRONICS ASSEM. INTO S/C STRUCTURE
11660
PREPARE TO SIMULATE MIDCOURSE & ORBIT TRIM ENGINE RESPONSES
11680
PREPARE TO SIMULATE INSERTION ROLL CONTROL RESPONSES
11700
PREPARE TO SIMULATE ORBIT INSERTION MOTOR RESPONSES
11720
PREPARE TO SIMULATE CAPSULE ELECTRICAL INTERFACE WITH SPACECRAFT
11740
PREPARE TO SIMULATE LAUNCH VEHICLE ELECTRICAL INTERFACE WITH SPACECRAFT
11760
PROVIDE TWO-AXIS ROTATION CAPABILITY FOR S/C STRUCTURE
11780
SIMULATE ZERO GRAVITY FOR ANTENNA ALIGNMENT
11800
SIMULATE SQUIBS TO SPACECRAFT
11810
DEPLOY SCAN PLATFORM
11820
SIMULATE CANOPUS SENSOR STIMULUS TO S/C
11840
SIMULATE PLANETARY SENSOR STIMULUS TO S/C
11860
SIMULATE SOLAR SENSOR STIMULUS TO S/C
11880
ACTIVATE ENGINE GIMBALLING SYSTEM
11900
PREPARE S/C FOR ELECTRICAL & ELECTRONIC S/S TEST
11950
SIMULATE A/C SYSTEM CONTROL VALVES TO SPACECRAFT

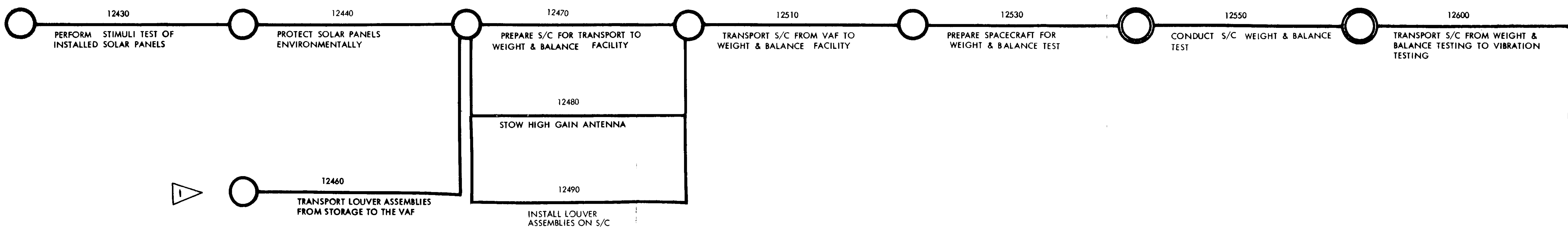


S/C COMPONENT ASSEMBLIES ARE ASSUMED TO HAVE BEEN FABRICATED AND CHECKED OUT PRIOR TO THIS FLOW SEQUENCE.

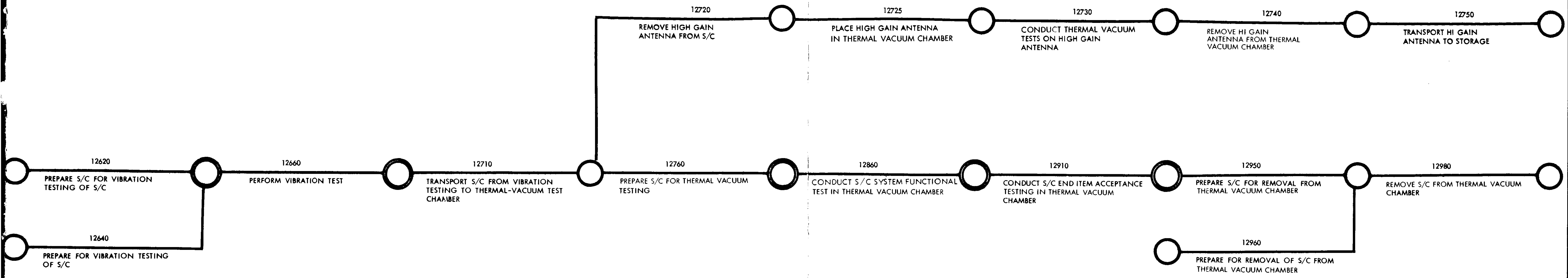
Rev.	Date	THE BOEING COMPANY			
			Initials	Date	VOYAGER SPACECRAFT
		Drawn			P.E.N. SERIES 11,000 thru 12,000
		Checked			
		Sys. Eng.			
		Super.			Assembly and Test of Spacecraft (S/C)
		Approved			
		Approved			

BOEING NO. FIG A -2
SH

REV LTR



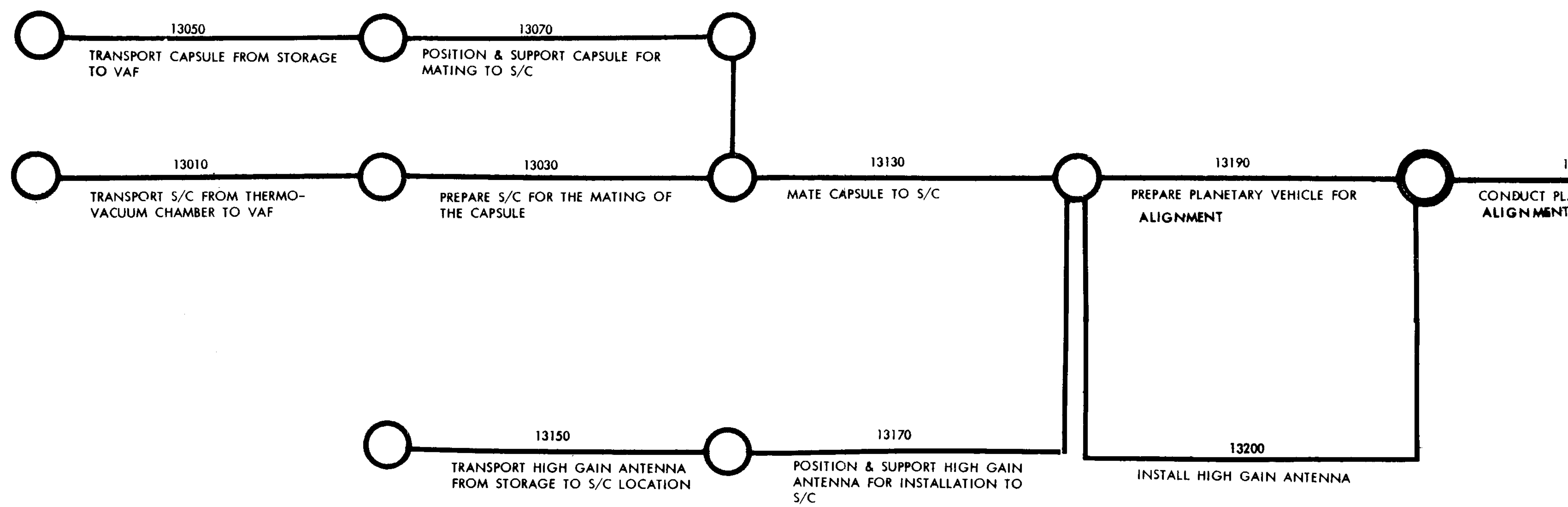
1 DUE TO FRAGILE NATURE OF LOUVER ASSEMBLYS A SPECIAL HANDLING PROCEDURE OR PROTECTIVE COVER ASSEMBLYS WILL BE REQUIRED DURING SUBSEQUENT TESTS.

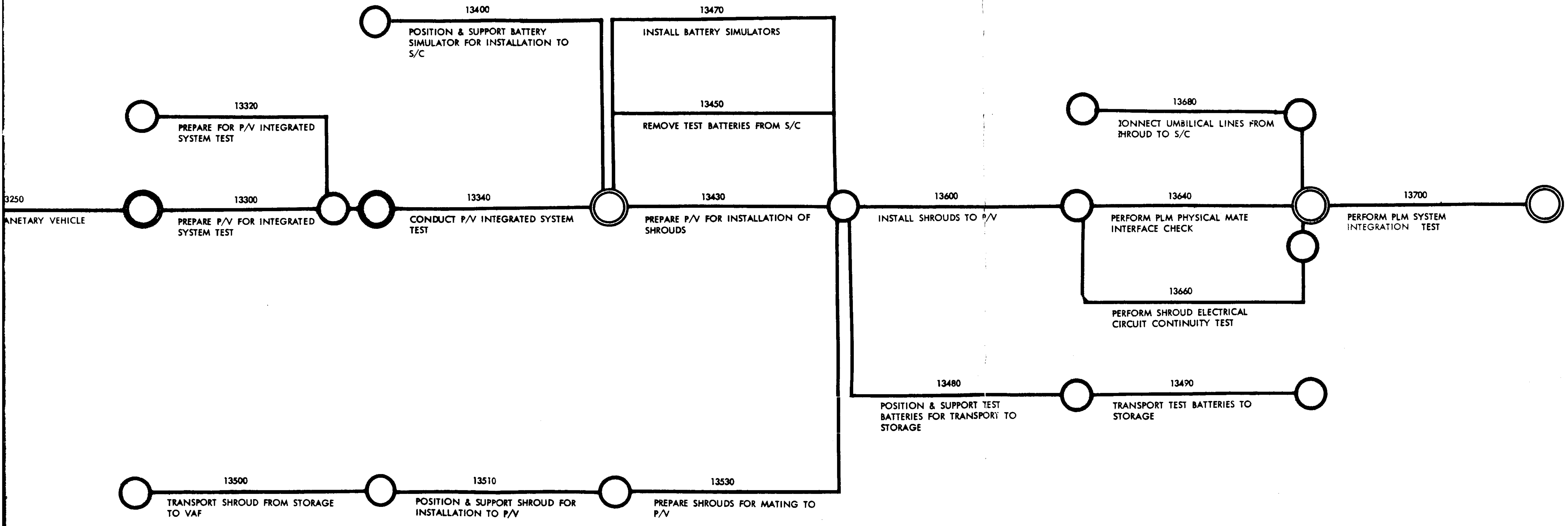


Rev.	Date	THE BOEING COMPANY			
			Initials	Date	VOYAGER SPACECRAFT
		Drawn			P.E.N. SERIES 11,000 thru 12,000
		Checked			
		Sys. Eng.			Assembly and Test of Spacecraft (S/C)
		Super.			
		Approved			
		Approved			

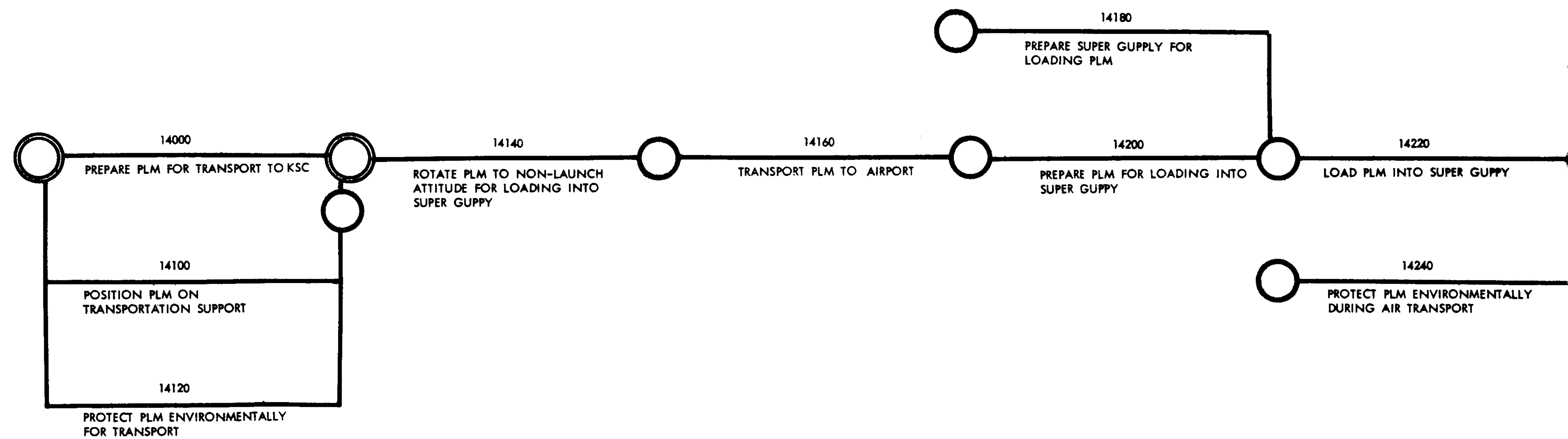
LDOUT FRAME 3 FOLDOUT FRAME FOLDOUT FRAME 4 REV TR FOLDOUT FRAME 5

BOEING NO. FIG A -3
SH

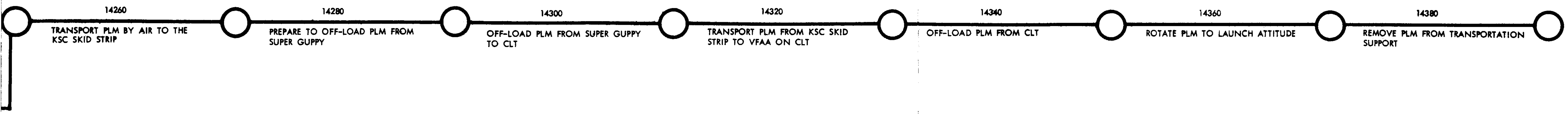

13050
TRANSPORT CAPSULE FROM STORAGE TO VAF
13070
POSITION & SUPPORT CAPSULE FOR MATING TO S/C
13010
TRANSPORT S/C FROM THERMO-VACUUM CHAMBER TO VAF
13030
PREPARE S/C FOR THE MATING OF THE CAPSULE
13130
MATE CAPSULE TO S/C
13190
PREPARE PLANETARY VEHICLE FOR ALIGNMENT
CONDUCT PL
ALIGNMENT
13150
TRANSPORT HIGH GAIN ANTENNA FROM STORAGE TO S/C LOCATION
13170
POSITION & SUPPORT HIGH GAIN ANTENNA FOR INSTALLATION TO S/C
13200
INSTALL HIGH GAIN ANTENNA



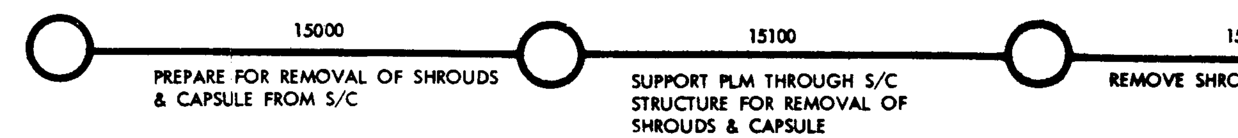
Rev.	Date				THE BOEING COMPANY
			Initials	Date	VOYAGER SPACECRAFT
		Drawn			P.E.N. SERIES 13,000 Assembly and Test Planetary Vehicle (P/V) and Payload Module (PLM)
		Checked			
		Sys. Eng.			
		Super.			
		Approved			
		Approved			

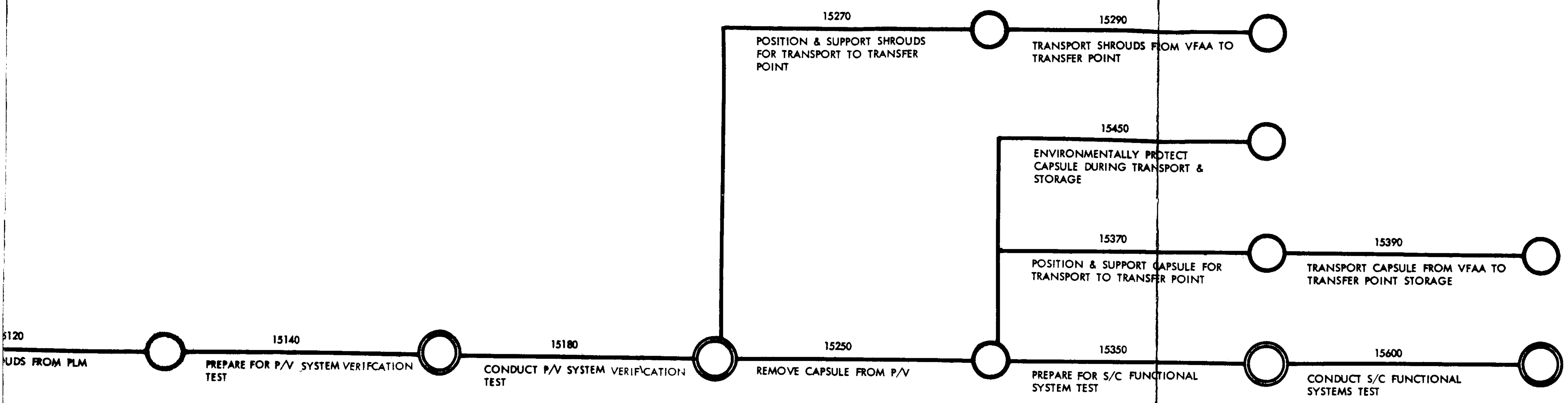


FOLDOUT FRAME FOLDOUT FRAME 1 FOLDOUT FRAME 1



Rev.	Date	THE BOEING COMPANY			
			Initials	Date	VOYAGER SPACECRAFT
		Drawn			P.E.N. SERIES 14,000
		Checked			Transport PLM to KSC
		Sys. Eng.			
		Super.			
		Approved			
		Approved			


15000
PREPARE FOR REMOVAL OF SHROUDS & CAPSULE FROM S/C
15100
SUPPORT PLM THROUGH S/C STRUCTURE FOR REMOVAL OF SHROUDS & CAPSULE



Rev.	Date	THE BOEING COMPANY			
			Initials	Date	VOYAGER SPACECRAFT
		Drawn			P.E.N. SERIES 15,000
		Checked			Disassemble PLM and Test S/C at KSC
		Sys. Eng.			
		Super.			
		Approved			
		Approved			

FOLDOUT FRAME 2

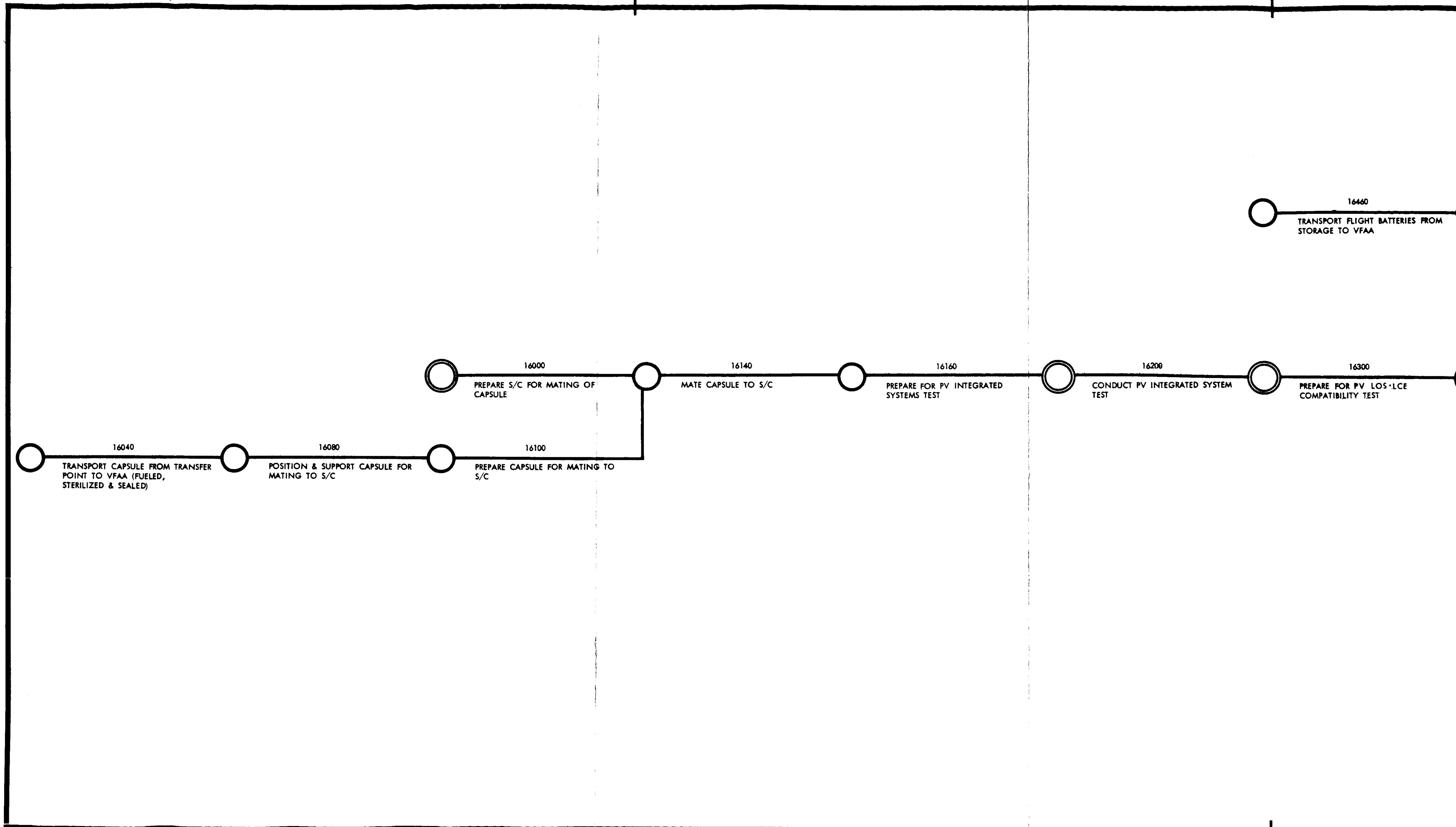
FOLDOUT FRAME 3

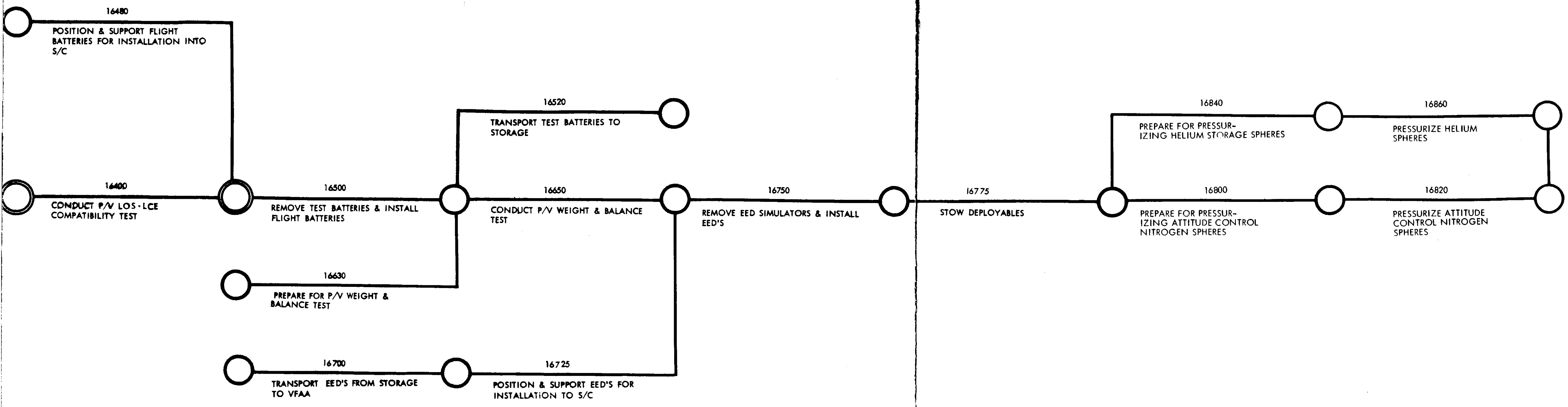
REV LTR

FOLDOUT FRAME 4

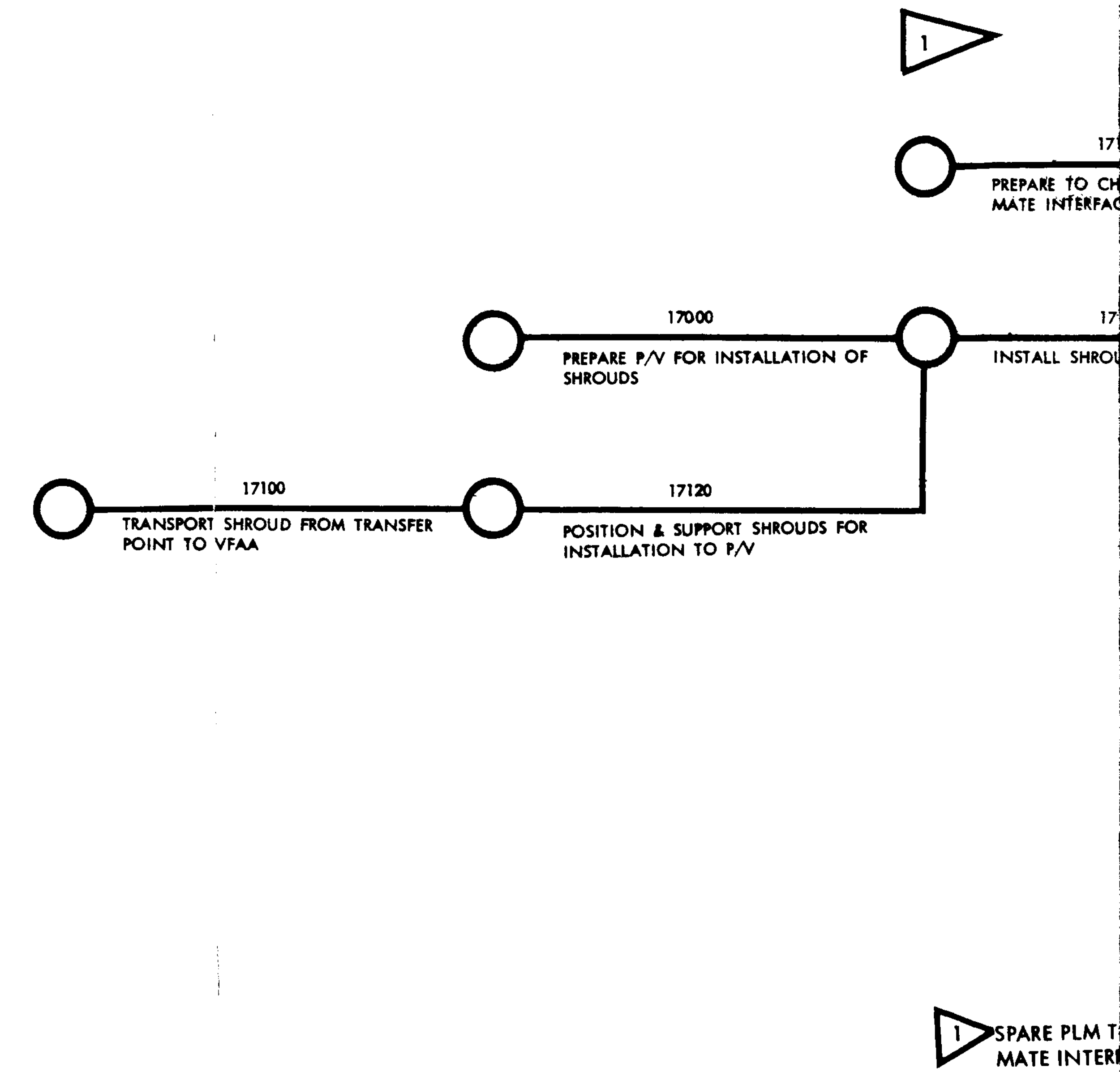
BOEING NO. FIG A -6

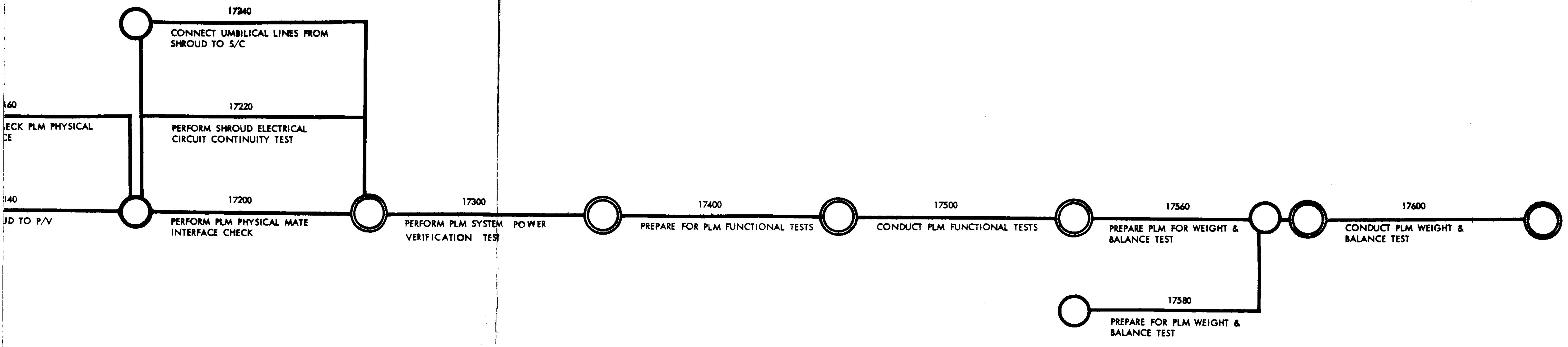
SH


16460
TRANSPORT FLIGHT BATTERIES FROM STORAGE TO VFAA
16000
PREPARE S/C FOR MATING OF CAPSULE
16140
MATE CAPSULE TO S/C
16160
PREPARE FOR PV INTEGRATED SYSTEMS TEST
16200
CONDUCT PV INTEGRATED SYSTEM TEST
16300
PREPARE FOR PV LOS-LCE COMPATIBILITY TEST
16040
TRANSPORT CAPSULE FROM TRANSFER POINT TO VFAA (FUELED, STERILIZED & SEALED)
16080
POSITION & SUPPORT CAPSULE FOR MATING TO S/C
16100
PREPARE CAPSULE FOR MATING TO S/C



Rev.	Date	THE BOEING COMPANY			
			Initials	Date	VOYAGER SPACECRAFT
		Drawn			P.E.N. SERIES 16,000
		Checked			Assemble Flight P/V and Test
		Sys. Eng.			
		Super.			
		Approved			
		Approved			


1
PREPARE TO CH
MATE INTERFAC
17000
PREPARE P/V FOR INSTALLATION OF SHROUDS
INSTALL SHROU
17100
TRANSPORT SHROUD FROM TRANSFER POINT TO VFAA
17120
POSITION & SUPPORT SHROUDS FOR INSTALLATION TO P/V
1
SPARE PLM T
MATE INTER
BOTH UPPER

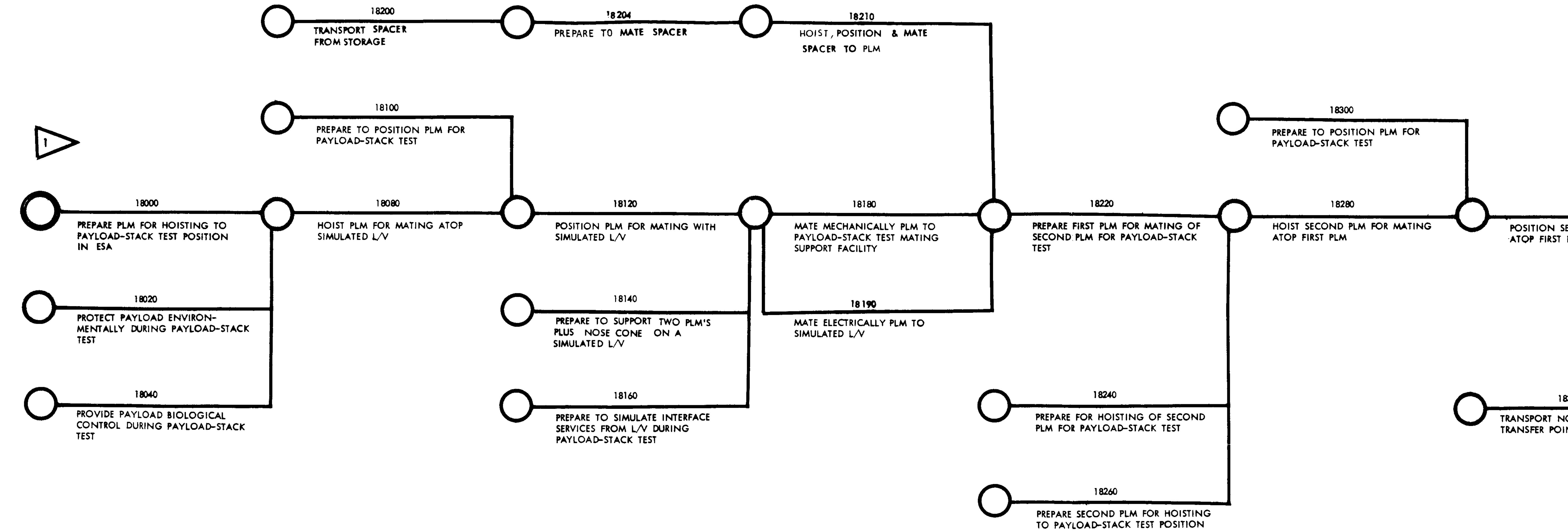


O BE PHYSICAL
ACE TESTED FOR
& LOWER POSITIONS

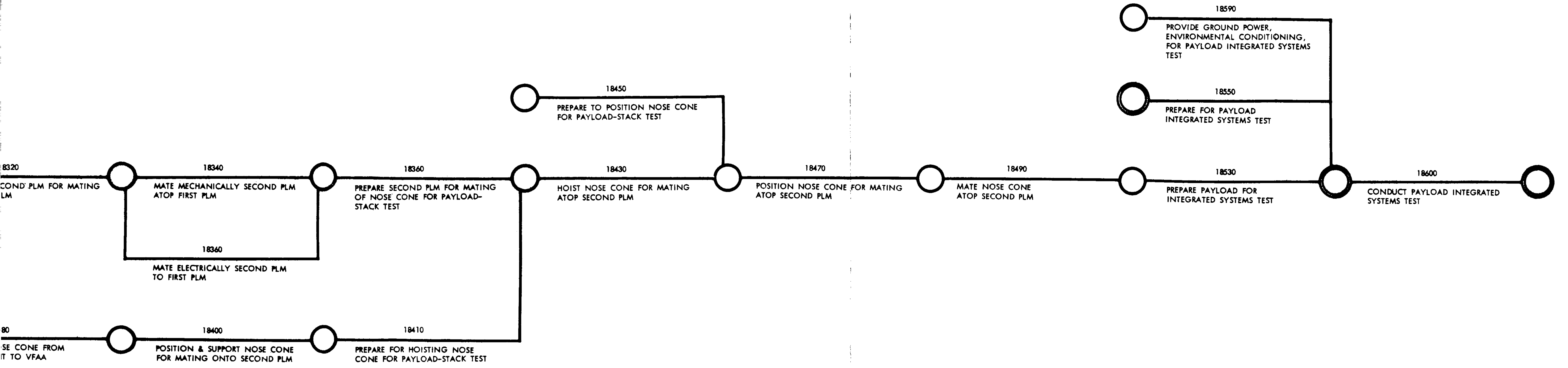
Rev.	Date	THE BOEING COMPANY		
			Initials	Date
		Drawn		
		Checked		
		Sys. Eng.		
		Super.		
		Approved		
		Approved		

VOYAGER SPACECRAFT

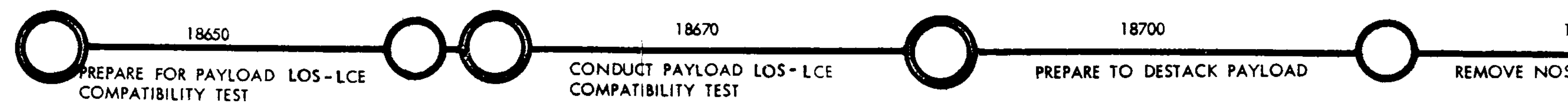
P.E.N. SERIES 17,000
Assemble Flight PLM and Test



1 FUNCTIONS 18,000 THROUGH 18,600 TO BE REPEATED TWICE FOR THIRD (SPARE) PLM. I.E., SPARE PLM TO BE MATED IN UPPER & LOWER POSITIONS.



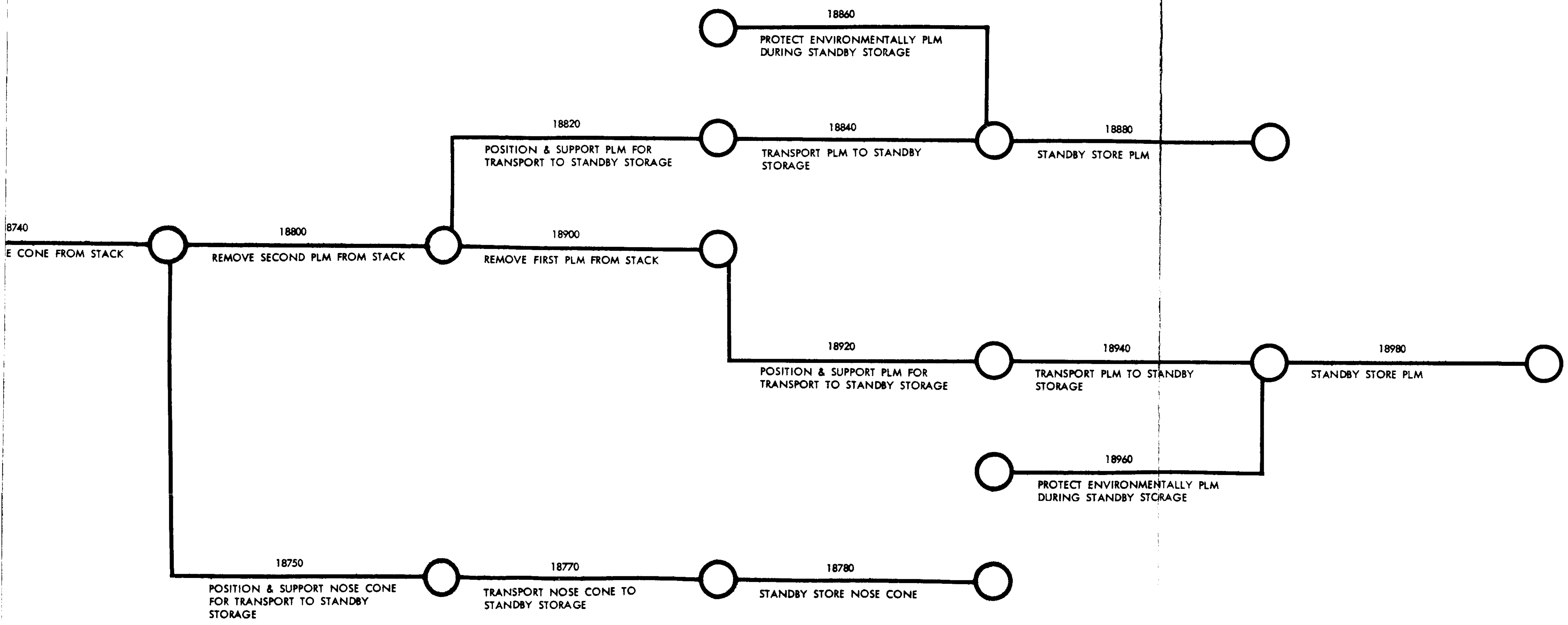
Rev.	Date			THE BOEING COMPANY
			Initials	Date	VOYAGER SPACECRAFT
		Drawn			P.E.N. SERIES 18,000
		Checked			Stack Flight Payload, Test,
		Sys. Eng.			Destack and Standby Storage
		Super.			
		Approved			
		Approved			



FOLDOUT FRAME

A-10

FOLDOUT FRAME /

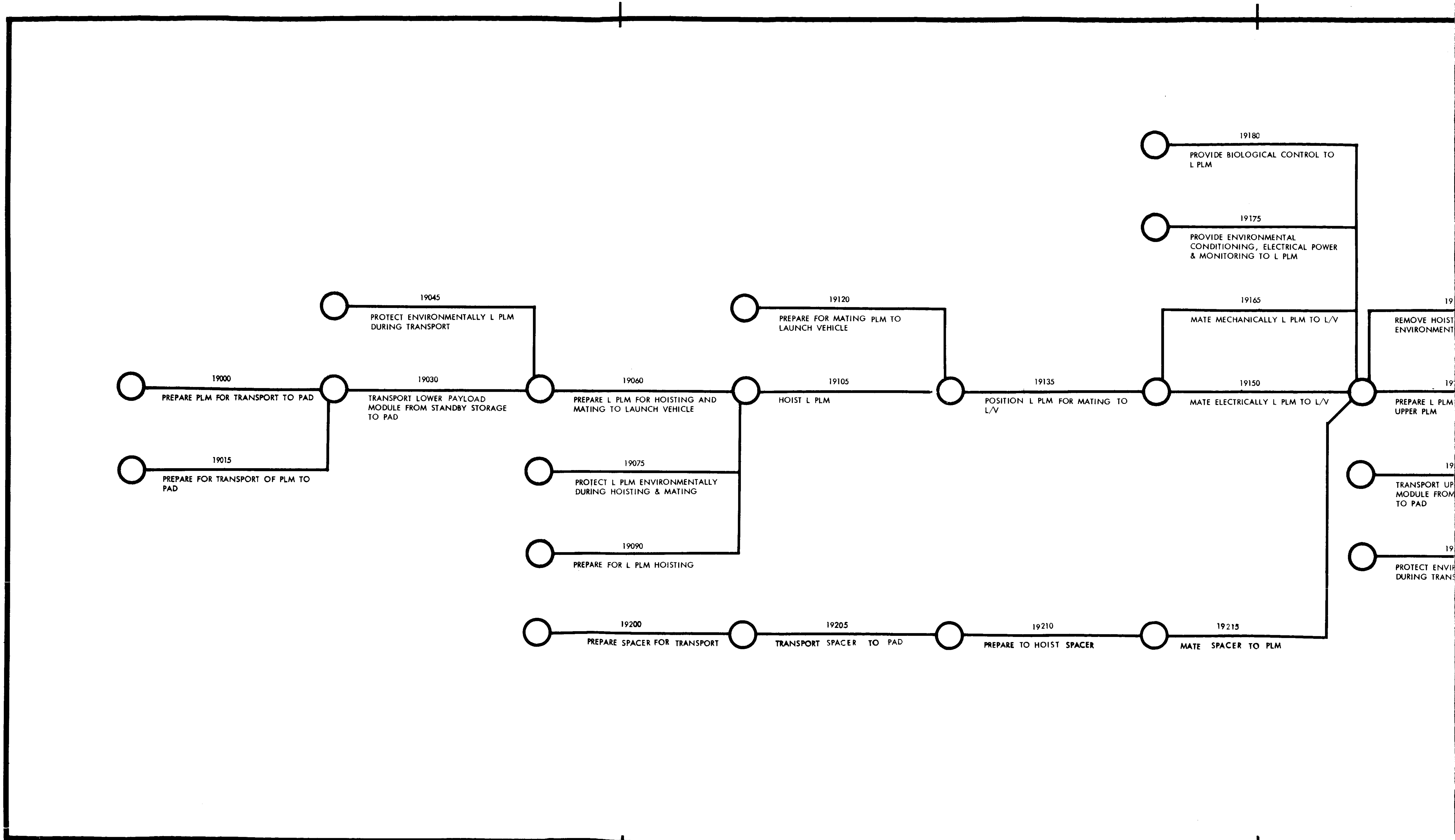


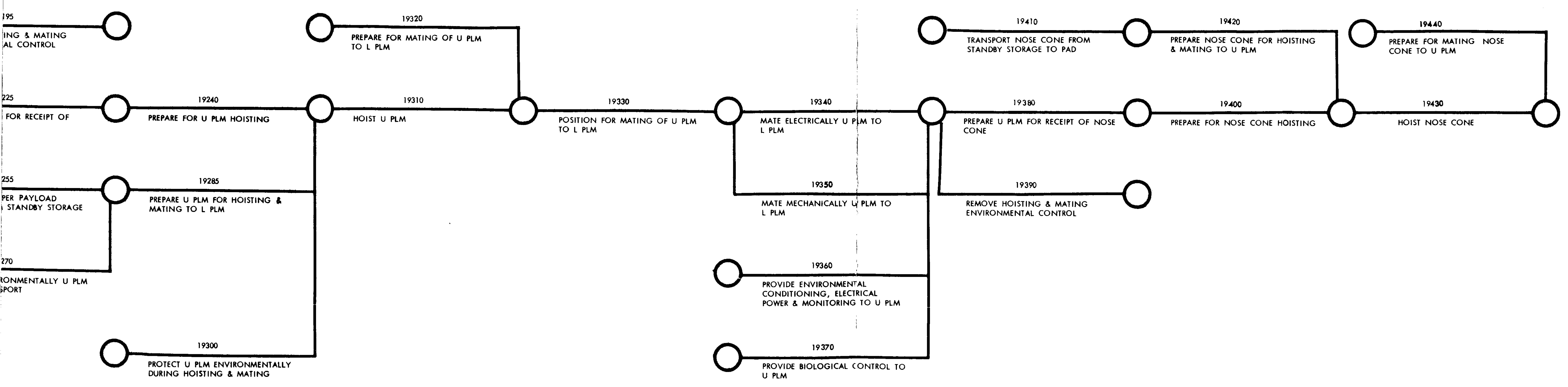
Rev.	Date	THE BOEING COMPANY			
			Initials	Date	VOYAGER SPACECRAFT
		Drawn			P.E.N. SERIES 18,000 Stack Flight Payload, Test, Destack and Standby Storage
		Checked			
		Sys. Eng.			
		Super.			
		Approved			
		Approved			

FOLDOUT FRAME 2 FOLDOUT FRAME FOLDOUT FRAME 3

REV LTR FOLDOUT FRAME 4 BOEING NO. FIG A -10

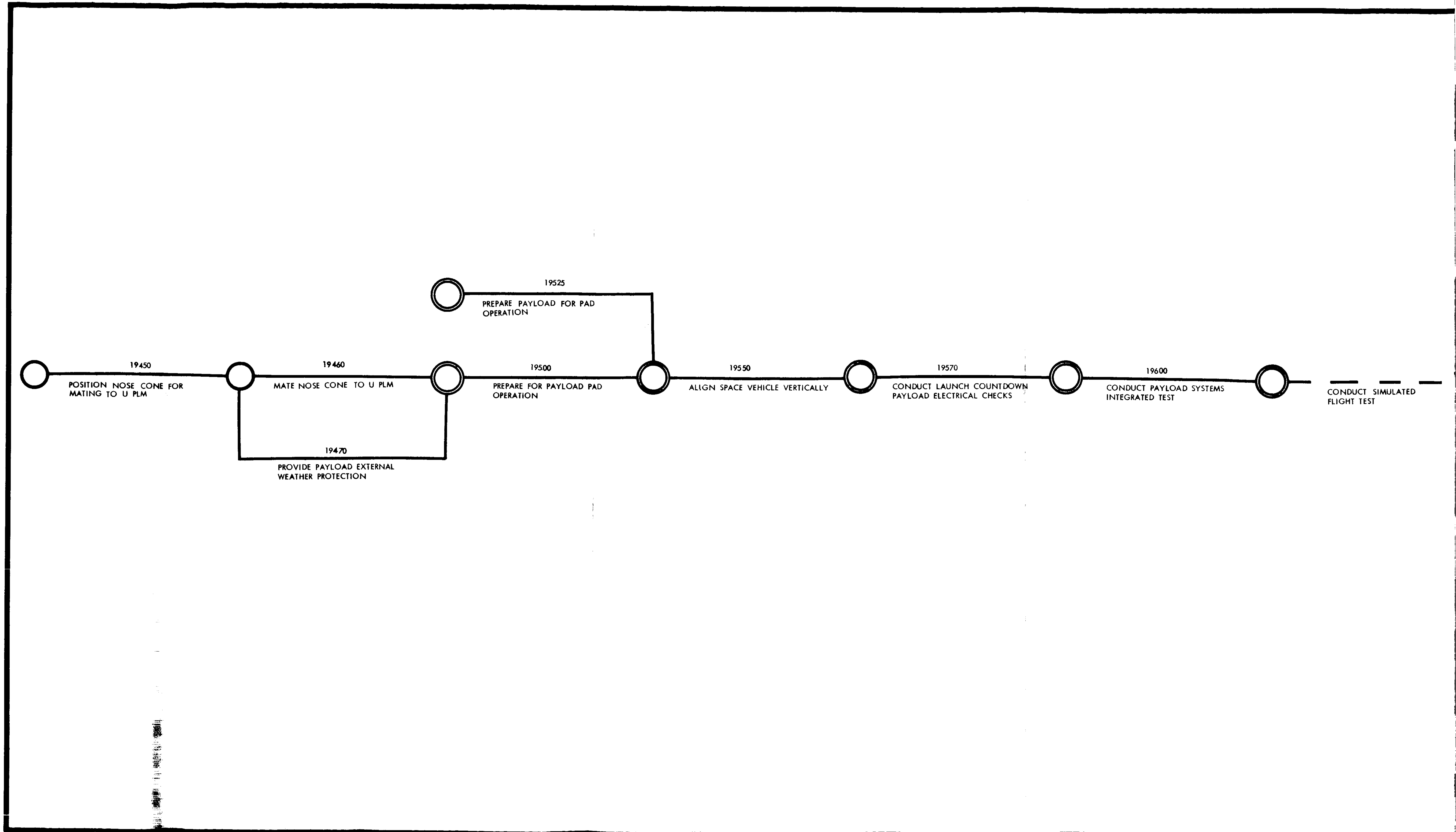
SH


19180
PROVIDE BIOLOGICAL CONTROL TO L PLM
19175
PROVIDE ENVIRONMENTAL CONDITIONING, ELECTRICAL POWER & MONITORING TO L PLM
19045
PROTECT ENVIRONMENTALLY L PLM DURING TRANSPORT
19120
PREPARE FOR MATING PLM TO LAUNCH VEHICLE
19165
MATE MECHANICALLY L PLM TO L/V
REMOVE HOIST
ENVIRONMENT
19000
PREPARE PLM FOR TRANSPORT TO PAD
19030
TRANSPORT LOWER PAYLOAD MODULE FROM STANDBY STORAGE TO PAD
19060
PREPARE L PLM FOR HOISTING AND MATING TO LAUNCH VEHICLE
19105
HOIST L PLM
19135
POSITION L PLM FOR MATING TO L/V
19150
MATE ELECTRICALLY L PLM TO L/V
PREPARE L PLM
UPPER PLM
19015
PREPARE FOR TRANSPORT OF PLM TO PAD
19075
PROTECT L PLM ENVIRONMENTALLY DURING HOISTING & MATING
TRANSPORT UP
MODULE FROM
TO PAD
19090
PREPARE FOR L PLM HOISTING
PROTECT ENVI
DURING TRAN
19200
PREPARE SPACER FOR TRANSPORT
19205
TRANSPORT SPACER TO PAD
19210
PREPARE TO HOIST SPACER
19215
MATE SPACER TO PLM



Rev.	Date	THE BOEING COMPANY		
			Initials	Date	VOYAGER SPACECRAFT
		Drawn		
		Checked		
		Sys. Eng.		
		Super.		
		Approved		
		Approved		

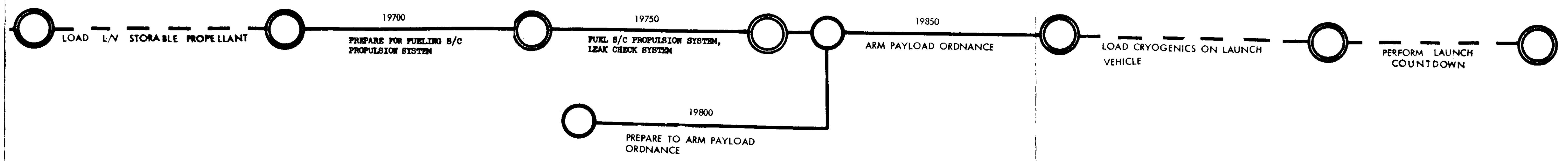
P.E.N. SERIES 19,000
Transport to Pad, Stack to Space Vehicle, Fuel S/C, Launch Countdown



FOLDOUT FRAME 1

FOLDOUT FRAME 2

FOLDOUT FRAME 2



Rev.	Date	THE BOEING COMPANY			
			Initials	Date	VOYAGER SPACECRAFT
		Drawn			P.E.N. SERIES 19,000 Transport to Pad, Stack to Space Vehicle, Fuel S/C, Launch Countdown
		Checked			
		Sys. Eng.			
		Super.			
		Approved			
		Approved			

FOLDOUT FRAME 3

FOLDOUT FRAME

FOLDOUT FRAME 4

FOLDOUT FRAME 5

FIG A -12

ERRATA SHEET

Page	Errata
3-19	Graph (a) change callout on $\dot{P}$ to read "0.52^{o}/day"
3-31	Figure 3.1-24 in the definitions, change second entry to read: "T perpendicular to S, and lying parallel to the ecliptic plane"
3-41	Figure 3.1-31 in the definition of the coordinate system, change the second entry to read: "$\overline{T}$ - normal to $\overline{S}$ and parallel to the plane of the ecliptic"
3-43	First line of fourth paragraph, change "convariance" to "covariance"
3-50	In the second line, change "Point" to "Pointing"
3-70	At the end of the eighth line of the second paragraph, delete "p"
3-91	In the second line of the second paragraph, change "1973-1979" to "1975-1979"
4-5 and 4-6	In Figure 4-2, in the upper part of the PLAN VIEW change "+Z AXIS" to "-Z AXIS"
	Below +Y axis in plan view change "(TYP. 4PL)" to "(TYP. 2 PL)" for propulsion louver assembly callout
5-2	In the fourth line of the second paragraph, change "5.4.1" to "5.4"
5-4	In the third line of the last paragraph, change "patch" to "path"
5-16	In item 40) change "Rate -0.02 degree - 0.2 deg/sec" to "Rate - 0.02 to 0.2 deg/sec)"